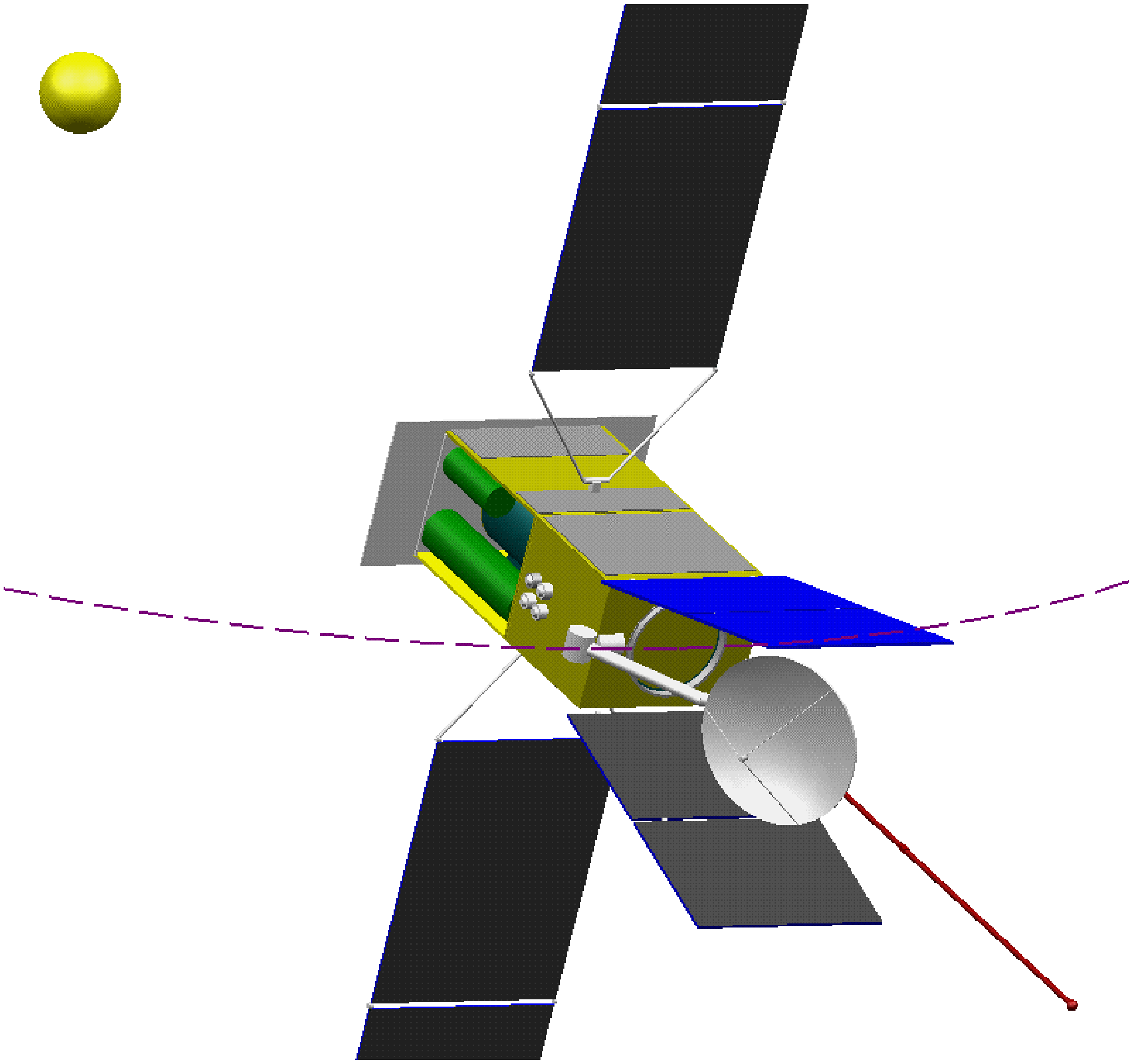
PRE-ASSESSMENT STUDY REPORT

SOLAR ORBITER



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1 INTRODUCTION

1.1 Background

In 1997, ESA formed the Solar Physics Planning Group (SPPG), in an effort to bring to a focus the European-wide discussion on future solar missions after SOHO. Following extensive discussions at a meeting arranged by the SPPG in Tenerife (23-27 March 1998; 'A Crossroads for European Solar and Heliospheric Physics', ESA SP-417, June 1998), a recommendation was formulated to the effect that a Solar Orbiter mission would provide the next major steps forward in our exploration of the Sun. Building on that recommendation, a further meeting was held at the Max Planck Institute for Aeronomie, Lindau, Germany (26-30 October 1998) which was aimed at providing a better focus on the scientific goals and orbiter mission concepts. An important aspect was how such a mission could be realised in the framework of an ESA 'flexible' mission (F) with a launch in the timeframe 2007-2009. ESA agreed to conduct a mission study for a Solar Orbiter at pre-assessment level, and a kick-off meeting with members of the SPPG took place at ESTEC on March 25th 1999.

1.2 Scope

The objective of the Solar Orbiter (SOLO) study was to examine the technical feasibility of a mission fulfilling the scientific requirements defined by the SPPG at a pre-assessment level, with particular emphasis on orbit design and spacecraft system aspects.

This report documents the study performed at ESTEC by an ESA team, using the ESTEC Concurrent Design Facility.

1.3 Document Structure

The Executive Summary (contained herein) provides a description of the scientific aims, instruments, spacecraft, launcher, mission exploitation and programmatics, which comprises the Solar Orbiter mission. Details of each domain addressed in the study are contained in subsequent chapters. A report on the analysis of the mission radiation environment is included in appendix. Costing information is published in a separate document (CDF-02(B)), due to the different distribution requirements. However, cost assumptions are given in this report, excluding any figures.

2 EXECUTIVE SUMMARY

2.1 Assumptions and Constraints

The study has been performed under the main assumption that the mission is completely financed by ESA, i.e. no international co-operation has been considered.

ESA would be responsible for

- The overall spacecraft and mission design, including the payload part of it.
- Payload elements accommodation and integration into the spacecraft bus, including appendages and antennae.
- System testing.
- Spacecraft launch and operations.
- Acquisition and distribution of data.

It should be noted that the payload element interfaces have been defined and relevant physical characteristics (mass, power, accommodation) have been accounted for at system level.

The mission cost has been targeted within the envelope of a Flexible mission, i.e. 175 MEuro, with programmatic issues modelled on Mars Express, in terms of industrial organisation, model philosophy, AIV approach, and launcher type.

Maximum use has been made of already developed (or to be developed and available in the year 2004) technology, mostly derived from the technology programme of Mercury, Cornerstone Nr 5 candidate mission.

The launcher is a standard Soyuz-Fregat, assumed launched from the Kourou launch pad. This assumption needs to be verified if the mission is approved. At present, Soyuz can only be launched only from Baikonur, but with much lower performance in term of lift-off mass (1310 kg instead of 1560 kg).

The spacecraft operations are performed by ESOC, using only one ground station and will have a nominal duration of 4.74 years, extendable to 7 years. The ground station assumed to be available is Perth, which needs to be confirmed. However, an antenna dish of minimum 35-m diameter, working in deep space Ka-band, will be mandatory to satisfy the scientific data rates to ground.

The design life of the spacecraft is compatible with the nominal lifetime, i.e. 4.74 years, whilst the consumables (i.e. fuel, solar array) have been dimensioned for the extended mission duration (7 year).

The scientific observation during the cruise phase of 1.86 years would be possible, subject to compatibility with the operation plan of the spacecraft. For example, observation would not be possible during the firing periods of the electrical propulsion.

Finally, the baseline orbit is the so-called Soleil10B_STP.

2.2 Mission & Spacecraft Summary

The following table gives an overview of the mission and the main characteristics of the Solar Orbiter spacecraft. Details can be found in the relevant chapters of this report.

Scientific Objective	View the Sun from an out-of-ecliptic, near-Sun, heliocentric orbit • Spectroscopy and imaging at high spatial and temporal resolution • In-situ sampling of particles and fields from a quasi-corotational perspective • Remote-sensing of the polar regions of the Sun
Payload	Instruments totalling 145 kg (174 kg inc. 20% margin) consuming 130 W power. Particle experiments: – Solar Wind Analyser – Plasma Wave Analyser – Particle Detector – Dust Detector Imaging and Spectrometry Package: – EUV/X-Ray Imager – EUV Spectrometer/Imager – Magnetograph – Coronagraph
Launcher	Dedicated launch with Soyuz-LV Fregat. Preferred launch site is Kourou (launcher payload 1560 kg).
Spacecraft	• Design Lifetime = 4.74 yrs, consumables sized for 7.01 yrs (extended mission) • Total mass = 1510kg • Main s/c bus: 3000 mm x 1200 mm x 1600 mm. • 3-Axis stabilised. • Solar Electric Propulsion system: 4 x 0.15N Stationary Plasma Thrusters. • Pointing Stability better than 3 arcsec/15min • Deployable and rotatable Cruise solar arrays, total 28 m^2, GaAs cells, jettisoned after last SEP thrusting. • Deployable and tiltable Orbit solar arrays, total 10 m^2, 16% GaAs cells, 84% OSR. • 5.8m Magnetometer boom shielded by the spacecraft body. • Four X-band LGAs, omni coverage, for TT&C. • One Ka-band HGA, 1.5m dia, for Telemetry after Cruise.

Cruise Phase	• Duration: 1.86 years • Five periods of thrust, between 1.21 and 0.33 AU. • Cruise solar arrays jettisoned after last firing.
Nominal Mission Phase	• Duration: 2.88 years • Initial Orbit: Perihelion 0.21 AU, Aphelion 0.9 AU, inclination 6.7 deg. • Final Orbit: Perihelion 0.3 AU, Aphelion 0.8 AU, inclination 23.4 deg. • Up to +/- 30 deg Solar Latitude. • High rate data acquisition during +/- 5 days perihelion, max/min latitudes. • Science download to Earth by HGA, above 0.5 AU orbit periods.
Extended Mission Phase	• Duration: 2.28 years • Final Orbit: Perihelion 0.3 AU, Aphelion 0.7 AU, inclination 31.7 deg. • 35.2 and – 38.3 Solar latitudes reached at end of phase.
Operations	• mission lifetime of 4.74 yrs (nominal), 7.01 yrs (extended) • Over 200 Gbits of data per orbit • LEOP perfomed by ESA using the Kourou ground station and ESOC flight control system; • Routine operations using the Perth 35m ground station linked to Solar Orbiter mission control centre
Programmatics	• Flexible Mission (F2/F3). • Assumed as an ESA financed mission. • Mercury Cornerstone heritage. • Target launch date: Jan 2009. • Development and life-cycle derived from Mars Express • Risk Analysis results are consistent with the proposed program approach.

3 SCIENTIFIC OBJECTIVES

3.1 Introduction

In recent years, our knowledge of the Sun, and its environment, the heliosphere, has increased dramatically, largely as a result of highly successful space missions like SOHO, Ulysses, Yohkoh, and TRACE. Nevertheless, fundamental questions remain, and significant progress towards answering these questions can be made by bringing instruments to as yet unexplored regions of the heliosphere. For example: we have never viewed the Sun directly from a solar orbiting platform; we have never viewed the Sun from within several tens of solar radii; we have never viewed the Sun directly from out of the ecliptic. Following extensive discussions within the scientific community, it was recommended that a Solar Orbiter mission, incorporating both a near-Sun and a high-latitude phase, would provide the next major step forward in our exploration of the Sun and heliosphere.

Ideally, the near-Sun phase of the mission should enable the Orbiter spacecraft to approach the Sun to within 30-40 solar radii during part of its orbit, thereby permitting observations from a quasi-heliosynchronous vantagepoint (so-called "co-rotation"). At these distances, the angular speed of a spacecraft near its perihelion approximately matches the rotation rate of the Sun, enabling instruments to track a given point on the Sun's surface for several days. On the other hand, during the out-of-ecliptic phase of the mission, the Orbiter ideally should reach solar latitudes of at least 40^{o}, so that the Sun’s polar caps are fully visible to the remote sensing instruments. A Solar Orbiter mission of this kind would enable unquestionable progress to be made in solving many of the fundamental problems remaining in solar and heliospheric science. In the following, a brief summary is given of the key scientific goals that could be achieved by the Solar Orbiter, addressing the near-Sun and out-of-ecliptic phases separately.

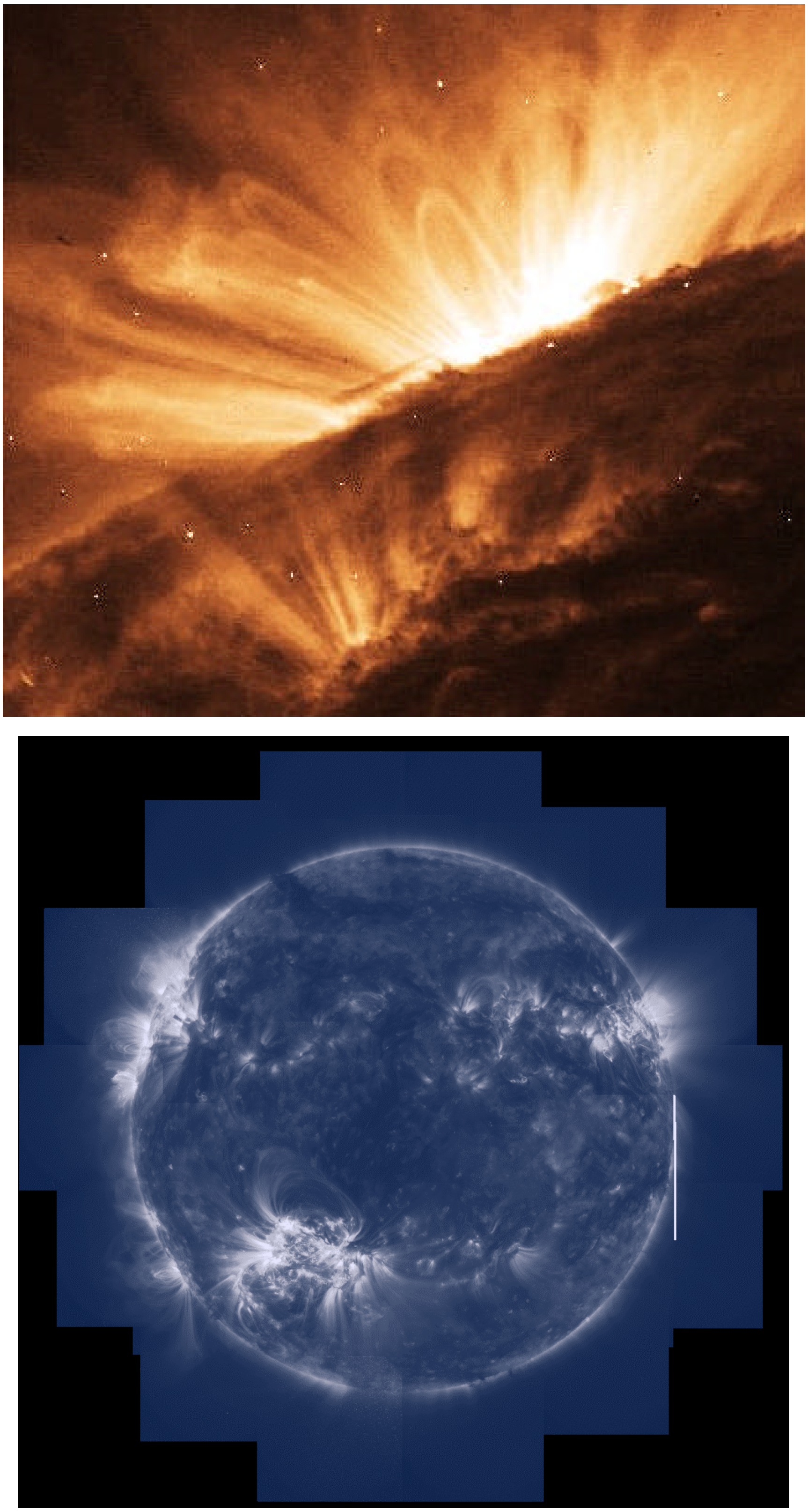


Figure 3-1: High-resolution image(s) of the Sun recorded by the NASA Transition Region and Coronal Explorer (TRACE) mission.

3.2 Goals for the near-Sun phase of the Solar Orbiter mission

The solar corona and the near-Sun solar wind within 60 solar radii are the last regions of the solar system to remain unexplored by in-situ measurements. The near-Sun phase of the Solar Orbiter mission is specifically designed to close these gaps of knowledge. In particular, the in-situ measurements should permit us to determine:

- The characteristics of plasma and electromagnetic fields within the quiet coronal streamer belt, and within coronal holes extending to the near-ecliptic regions (slow- and high-speed solar wind), as well as the characteristics of extended coronal mass ejections (CMEs) near the Sun to establish the differences in their sources and nature;

- The characteristics of energetic particles close to the Sun to understand their origin, acceleration and transport processes in the corona and inner heliosphere;
- The characteristics of the near-Sun dust and its origin and spatial distribution;
- The characteristics of coronal radio emissions.

On the other hand, the remote-sensing measurements of the solar atmosphere from a near-Sun perspective should enable us to better describe and understand:

- The nature and fine-scale structure of the source regions of the solar wind and of the coronal heating processes at the coronal base;
- The nature of disturbances associated with small-scale magnetic activity, flares (shocks) and eruptive prominences.

The Solar Orbiter mission as envisaged will permit us to correlate the in-situ observations with magnetic activity occurring near the solar surface in unprecedented detail. The quasi-heliosynchronous phase of the orbit near perihelion will allow us to observe active regions continuously, in particular their dynamics and evolution at the surface and in the atmosphere of the Sun, and to measure the interplanetary consequences at distances beyond 40 solar radii.

3.3 Goals for the out-of-ecliptic phase of the Solar Orbiter mission

Building on our knowledge and experience gained with Ulysses (which carries no imaging / spectroscopic instruments), and SOHO (which remains in the ecliptic plane), we can formulate a number of key objectives for the out-of-ecliptic phase of a Solar Orbiter mission. These include

- The investigation of the nature and evolution of the solar polar coronal holes and the origin of solar high speed wind streams;
- The investigation of coronal mass ejection global distribution, onset processes and propagation;
- The investigation of the Sun's magnetic field structure and evolution, in particular near the poles;
- The investigation of the dynamics and rotation of the solar poles;
- The investigation of the true irradiance variability of the Sun, for the first time.

Secondary goals include a unique understanding of coronal loops, and a detailed observation of several features of solar polar structure and activity. Observations from a spacecraft in a Sun-centred, inclined orbit, in combination with data from one or more in-ecliptic satellites, would provide a unique capability in the field of Space Weather. Obvious examples include the ability to reconstruct the three-dimensional structure of coronal mass ejections in interplanetary space, and the possibility to view all solar longitudes simultaneously.

3.4 Scientific instruments

A payload mass of under 100 kg is anticipated, split between a particles and fields package, and a remote-sensing package. This will represent a well-tuned mix of instrumentation for the investigation of source and in-situ plasmas. The remote sensing package should include a suitable imaging and spectroscopy capability. The combination of observations at relatively close proximity to the Sun, and high resolution (spatial, spectral and temporal) spectrometer and imager instrumentation will provide a major step forward. A spatial resolution of 1 arcsecond, and a temporal resolution of 1 second should be realistic aims, although the mission will demand significant miniaturisation of existing instrument designs. The package should include a high-resolution extreme UV (EUV) imager, a high resolution EUV spectrometer, a high resolution magnetograph and a coronagraph system. We note that the magnetograph can be used for helioseismological observations of the poles (see goals).

The particle and fields package should comprise a standard suite of instruments for monitoring the solar wind electron and ion distributions, energetic particles of solar and interplanetary origin, and heliospheric electric and magnetic fields. Other in-situ devices could include a radio experiment for coronal sounding and radio tomography, and a dust detector for measuring interplanetary dust particles.

The payload must be designed to cope with the extremes in temperature, which a relatively close approach to the Sun will provide. In addition, a radiation environment much worse than that encountered, for example, on board SOHO may be anticipated. An important consideration is that the bulk of payload operations must be highly autonomous. The restrictions placed on the downlink by the near-Sun trajectory (e.g., the need to stow the high-gain antenna near perihelion) result in a data acquisition strategy that involves on-board storage of many days' data prior to transmission to Earth. Daily "quick-look" operations will probably not be possible.

4 PAYLOAD REQUIREMENTS AND DEFINITION

4.1 Introduction

The Solar Orbiter model payload is aiming at spectroscopy and imaging at high spatial and temporal resolution, in-situ sampling of particles and fields from a quasi-corotational perspective and remote sensing of the polar regions of the sun. The model payload consists of a spectro-imager instrument package and of a particle and wave package (Additional information about the experiments can be found in the URL's: *http://solg2.bnsc.rl.ac.uk/~harrison/payload*).

A brief description of the payload instruments is provided below.

The spectro-imager package consists of the:

EUV/X-ray imager, imaging in one or more narrow spectral bands. A rotating filter mask is selecting the imaging spectral region. The instrument contains a shutter door for optics protection during the non-operational mode. The imaging resolution is typically 1 arcsec/pixel. The pointing stability requirement is 1 arcsec over 15 minutes and is a driver for the AOCS system. The detector has to be cooled to –80C, which poses specific requirements on the thermal system. This instrument will make use of the heritage from instruments flown on SOHO, YOKOH and TRACE.

EUV spectro-imager, which will take high resolution spectrally resolved partial images of the sun. The pointing stability requirement is 1 arcsec over 15 minutes. The detector has to be cooled to –80C. This instrument will make use of the heritage from instruments flown on SOHO and HALE.

Magnetograph, imaging in different wavelength ranges at very high resolution the sun (i.e. 0.5 arcsec/pixel). It incorporates its own image stabilisation system down to 0.01 arcsec stability. The pointing stability requirement is 1 arcsec over 15 minutes. The detector has to be cooled to –80C. This instrument will make use of the heritage from instruments flown on SOHO and HALE.

Coronograph, which is a traditional single white-light coronograph operating over a distance range from the sun of 1.5 to 20 solar radii and with a pixel resolution of 5 arcsec. The pointing stability requirement is 1 arcsec over 15 minutes. The detector has to be cooled to –80C. This instrument will make use of the heritage from instruments flown on SOHO.

The particle and wave package consists of the:

Solar wind analyser, pointing towards the sun and measuring ions (0-30 keV) and electrons (0-10 keV) velocity distribution, mass and charge.

Plasma wave analyser, employing an aerial boom system (tbd).

Magnetometer, (0.1 nT to 1 microT range) employing a boom (tbd).

Energetic particle detector, measuring ions and electrons pitch-angle and energy distribution.

Radio experiment, consisting of a sweep spectrometer with 4 channels of 0.1-2, 2-20, 20-200 and 200-2000 MHz employing an antenna system (tbd).

Dust detector, measuring interplanetary dust particles with masses in the range of 10E-16 g to 10E-6 g within a velocity range of 1 to more than 70 km/s.

The main payload resource requirements to the spacecraft are summarised below:

Experiment	Mass [kg]	Dimensions [m]	Power [W]	Data rate [kbit/s]	FOV [arcmin]	Pointing stability [arcsec/15 min]
EUV/X-ray imager	15	1.5x0.4 (diam)	20	8	60x60	1
EUV spectro-imager	50	2.8x0.4 (diam)	30	20	8x8	1
Magnetograph	40	1x0.4x0.25	35	25	30x30	1
Coronograph	25	0.75x0.3 (diam)	20	10	600x600	1
Particles and field package (Solar wind, Plasma wave, Magnetometer Particle detector	9	0.5x0.5x0.5 (total volume)	11	10	600x600 na na na	na
Radio experiment	5	0.2x0.2x0.2	10	1	na	na
Dust detector	1	0.1x0.1x0.1	1	0.5	1200x1200	na
Total	**145**	**na**	**127**	**74.5**	**na**	**1**

Table 4-1: Payload Summary

4.2 Payload complement effect on mission and spacecraft system

The spectro- imager package requires a three-axis stabilised platform. These instruments have to point to the sun with a very high pointing stability (1 arcsec/15 minutes) during science data acquisition. The absolute pointing accuracy requirement satisfying the needs of all instruments is better than 2 arcmin. The instruments have to be protected from sun illumination (except for the telescope apertures) and the instrument thermal system has to ensure –80 C CCD detector operating temperature.

Due to the high data rate generated during scientific operation, the spacecraft must provide sufficient storage memory to ensure three 10 day periods of operation per orbit for the high data rate experiments (at perihelion, northern maximum latitude and southern maximum latitude). The low data rate instruments shall be operable over the full orbit (pending occasional data storage and downlink limitations). The communication system has to be able to ensure the downlinking of this data to earth.

All deployment mechanisms and deployment monitoring devices (pyros, deployment mechanisms/sensors and motors) are part of the experiments. The commanding of the deployment and the deployment status monitoring is performed by the spacecraft. At the present moment the configuration of the wave package antennas and booms is not defined. The achievement of a magnetic cleanliness compatible with the magnetometer sensitivity requirement has to be studied in detail in the next phase.

The thermal design of the experiments is part of the respective designs. The spacecraft system will ensure that the instruments protected from sun illumination.

The power interface consists of a single 28 V regulated bus provided to the experiments. Power conditioning (including the generation of additional voltages) is under the responsibility of the experiments.

The commanding, the control and the acquisition of the scientific data is performed via direct digital I/O lines, analogue I/O lines and by 1553 or RS422 serial data bus all connected to the spacecraft data handling system. The exact number of the digital and analogue I/O lines is not yet fixed.

5 LAUNCH VEHICLES

The Solar Orbiter will be launched by a standard Soyuz/Fregat launch vehicle. The three stages Soyuz bring the nose module consisting of the upper stage with the spacecraft attached to it to an altitude of about 200Km, where the Fregat will circularise the orbit with a first burn. After a coast phase of about 70 minutes in the low earth orbit Fregat will be re-ignited for the boost into escape trajectory.

After separation and after the spacecraft has acquired Sun and deployed its solar arrays, spacecraft and payloads commissioning starts.

The main launcher parameters follow.

- Infinite velocity 2.44 km/s
- ΔV 4.8 km/s
- Maximum satellite mass at lift-off with SEP:
  - 1310 kg from Baikonur (including adapter mass)
  - 1560 kg from Kourou (including adapter mass)

The launch from Baikonur cosmodrome is carried out on azimuths corresponding to the inclination of 51.8°. A launch from Kourou would be performed at the inclination of 5°. The launch from Kourou is the assumed baseline for this mission due to payload mass reasons.

The satellite mechanical configuration is driven by the stringent mass and envelope constraints of the launcher. The definition of the interface between the spacecraft and the launch vehicle will require a close co-operation between ESA and STARSEM. However, based on preliminary estimates, CDF study Team proposes for the Solar Orbiter project to use an adapter based on the same concept as the one proposed for Mars Express.

The adapter design will be optimised in order to accommodate the spacecraft taking into account following constraints:

- satellite geometry, mass, structural loads requirements, etc.
- launcher usable volume and necessary fairing clearances, frequency requirements, and in certain respect operational constraints for satellite adapter integration.

Adapter weight is estimated around 50 kg.

Safe satellite separation is ensured through specific hardware and studies including:

- highly reliable design of the separation system itself through a fully redundant architecture (command, electrical link, initiators of the separation devices)
- mission analysis carried out to define the best strategy for spacecraft separation including Fregat avoidance manoeuvres with separated spacecraft and non-collision analysis of all bodies.

Figures of spacecraft mechanical configuration inside fairing are shown in the configuration chapter.

Other Launchers could be considered as options within the cost constraints:

- Long-March CZ-3A (1504 kg)
- Long-March CZ-4 (1648 kg)

Payload mass is calculated taking into account the infinite velocity requested for the present mission (2.44 km/sec).

6 MISSION ANALYSIS

The main Solar Orbiter scientific requirements for the orbit are:

- Low perihelion allowing co-rotation phases
- High inclination with respect to the Solar equator (at least 40°)
- Aphelion not higher than the Venus heliocentric distance

A trajectory design satisfying these requirements leads to high *Dv* and/or long transfer times unless low-thrust propulsion and gravity assist with planets is used. The baseline design:

- Is based on the same approach (by Y. Langevin) used in the Mercury mission trajectory design
- Needs only 1.86 y to reach an acceptable science orbit (0.89x0.21 AU) with
  - a perihelion of 45 SR
  - an orbit period of 149 days
  - an inclination ranging from 6.7° to 23.4° on the ecliptic (13.2° to 30.0° highest latitude w.r.t. Sun)

- With 0.3 N SEP thrust (specific impulse: 2100 s):
  - Total Δv = 4.77 km/s (nominal mission)
  - Total thrust time = 6033 h

- Seven perihelion passages at pseudo-synchronous viewing of 10 days
- Seven maximum Northern and Southern latitude viewing of 10 days each
- The orbit is resonant (2:3) with Venus so that each swing-by increases the inclination of the orbit; perihelion increases to 0.27 AU (58 SR)

The trajectory is composed of three phases:

1. Cruise phase, starting at spacecraft separation from the launcher, ending at start of scientific operations (some science may be performed during the cruise phase)

2. Nominal mission phase, during which the scientific mission is performed

3. Extended mission phase, if funding is available, the mission is extended and further gravity assist manoeuvres allow to better meet the requirement on high inclination.

During the cruise phase (0 - 1.86 y):

- 5 thrust phases; duration ranges from 6 to 105 days
- Venus swing-by's (for semi-major axis change) and inclination increase
- Perihelion passages at 0.33 AU (thrust phase) and 0.25 AU

During the nominal mission (1.86 - 4.74 y, duration: 2.88 y), there are 2 Venus swing-by's for inclination increase during 7 orbits:

Orbit	**1, 2, 3**	**4, 5, 6**	**7**
Maximum Solar latitude:	13°	22°	30°
perihelion passages at (AU):	0.21	0.23	0.27
Orbital rate at perihelion (°/day):	13.1	10.9	8.5

Table 6-1: Nominal Mission Orbit Characteristics

During the extended mission (4.74 - 7.01 y, duration: 2.28 y), there are 2 Venus swing-by's for inclination increase during 6 orbits:

Orbit	**8, 9**	**10, 11, 12**	**13**
Maximum Solar latitude:	30°	35°	38°
perihelion passages at (AU):	0.27	0.32	0.36
Orbital rate at perihelion (°/day):	8.5	6.5	5.3

Table 6-2: Extended Mission Orbit Characteristics

The ecliptic projection of the Solar Orbiter trajectory is shown in Figure 6-1.

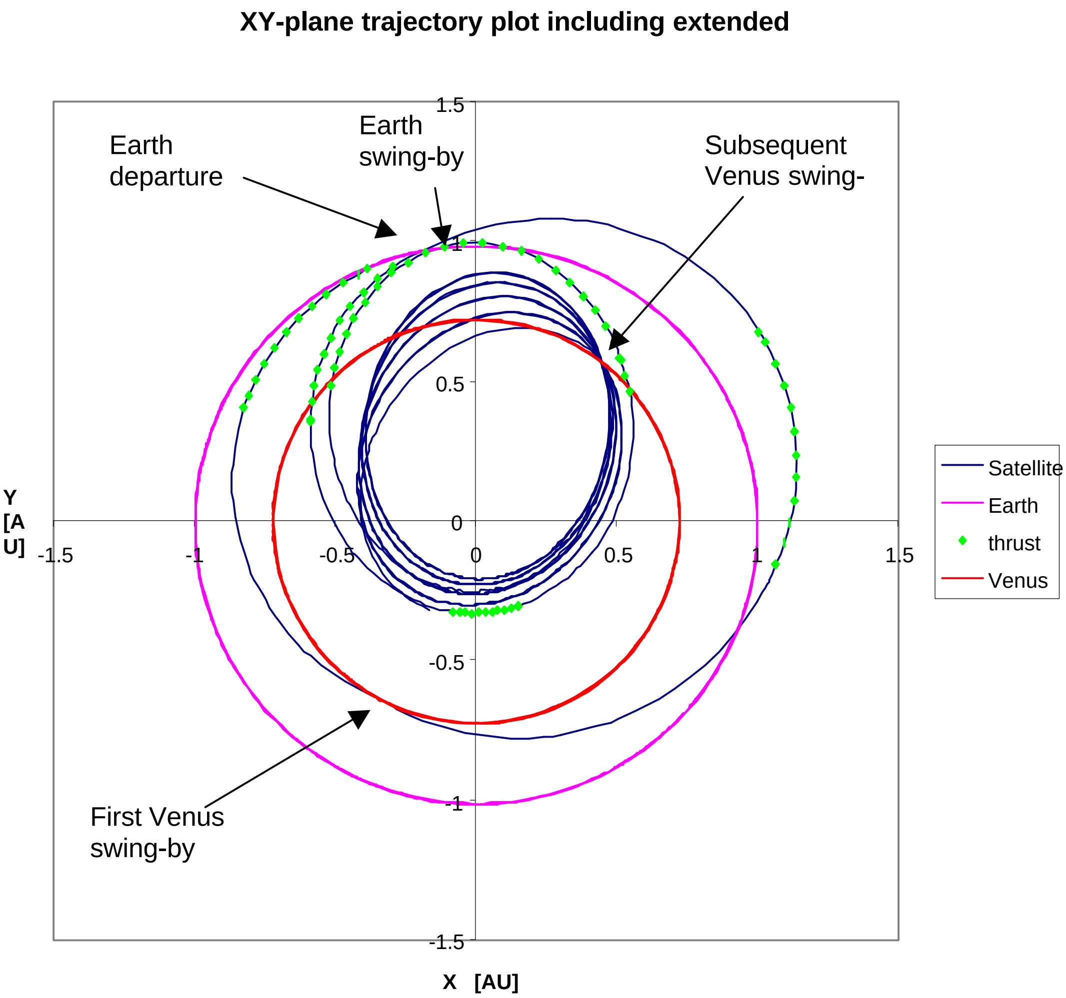


Figure 6-1: Ecliptic projection of the Solar Orbiter trajectory

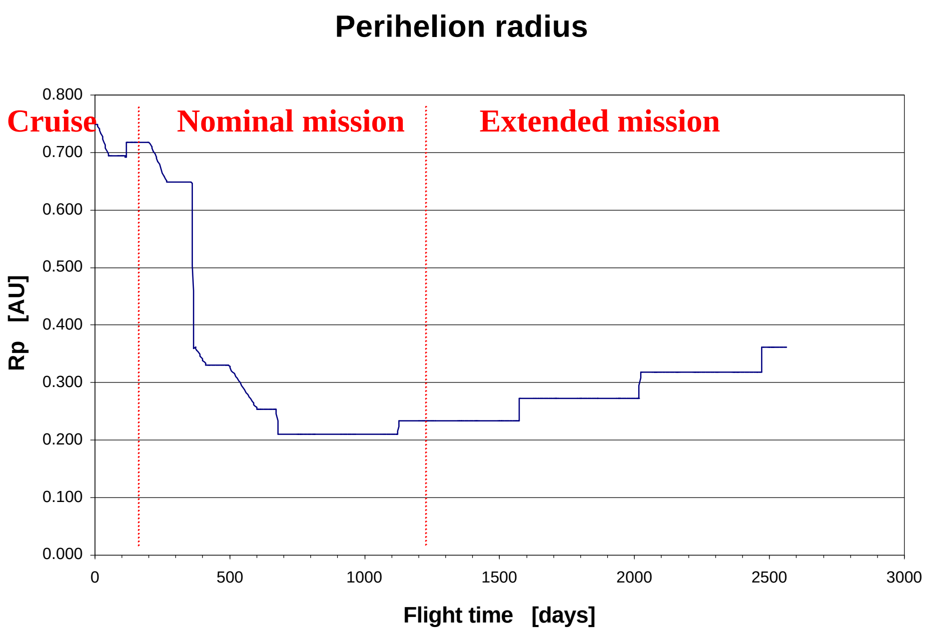


Figure 6-2: Perihelion distance as function of time

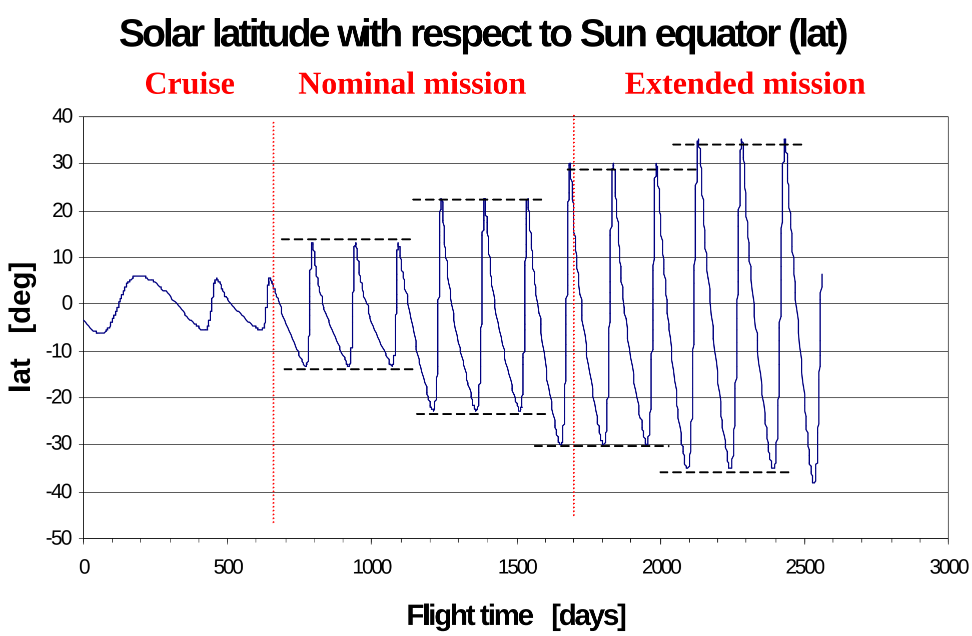


Figure 6-3: Solar latitiude as function of the flight time

7 SYSTEMS

7.1 System Design Drivers

The system design of the Solar Orbiter is driven by the Mission Requirements as well as by the general requirements and objectives of the Flexible Mission programme. The spacecraft is designed to get as close as possible to the Sun (0.21 AU) as the materials and engineering will allow. Figure 7-1 illustrates the maximum heat load, which the Solar Obiter and two other spacecraft will be subjected to during their missions. The orbit is highly inclined to the Sun's equator to the extent that the launcher and propulsion capability will allow. The power demand is higher during cruise, the majority being required by the electric propulsion system. The solar array sizing is compatible with the need to provide adequate power at the furthest distance from the Sun. The cruise solar array is a thermal burden to the system: when propulsion is no longer needed it is desirable to jettison. The spacecraft size and shape is directly linked to those of the instruments and the service module performance.

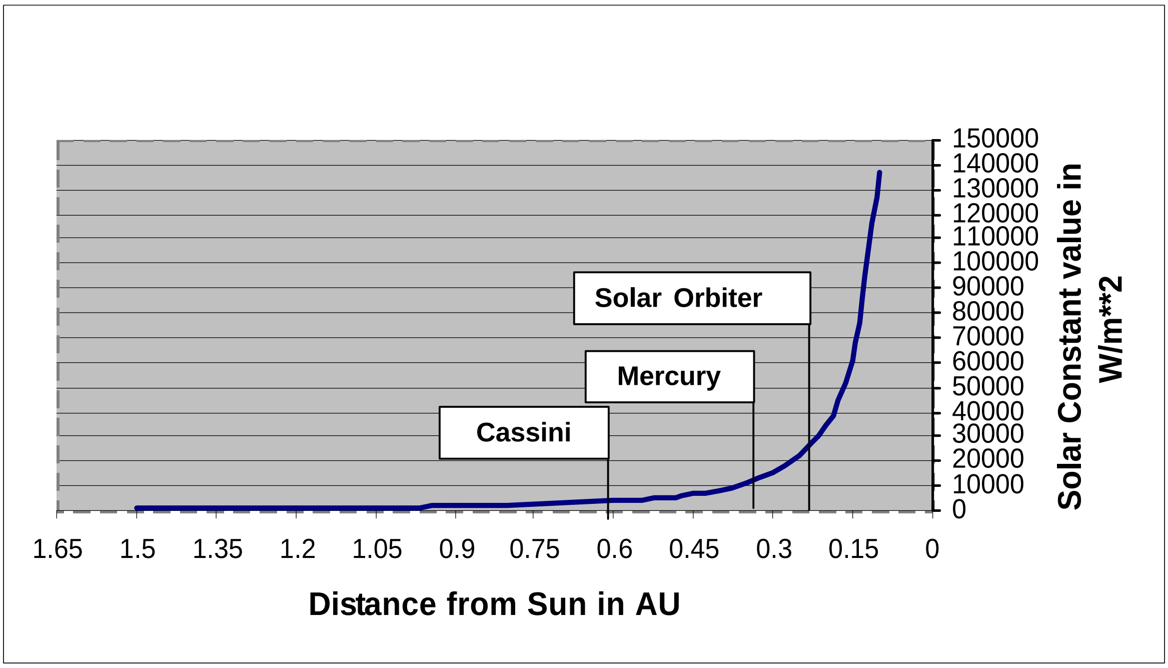


Figure 7-1: Solar Constant behaviour vs Distance from Sun

The major design driving features are:

- Instruments requirements, mainly field of view, pointing requirement stability, operations and size.
- Launcher mass envelope and interfaces
- Earth Communication, mainly accommodation of HGA, and HGA pointing for data transmission
- Use of existing hardware to minimise risk

The spacecraft functional architecture has been made maximises the use of hardware expected to be available by 2004, including the new technologies planned for Mercury.

7.2 Main System Design Features

The spacecraft has a box like configuration, approximately 3.0m long, 1.6m wide and 1.2m deep. It has internally mounted instruments, 2-axis steerable HGA, two sets of steerable (one degree of freedom) solar arrays and Electrical Propulsion based on 4*0.15N SPT thrusters mounted on +Z face of the spacecraft (See Configuration chapter for definition of spacecraft reference system).

The most demanding areas for the spacecraft design are described below:

7.2.1 Thermal Design

Proximity to the Sun is particularly demanding for the appendages, which cannot be protected by thermal shielding. The solar array and high gain antenna cannot sustain the temperatures to be encountered during the data collection phases. Particular measures are needed to reduce their exposure.

Figure 7-2 shows the temperature profile (perpendicular surface to the sun is considered) with distance from the sun using different technology for external coating.

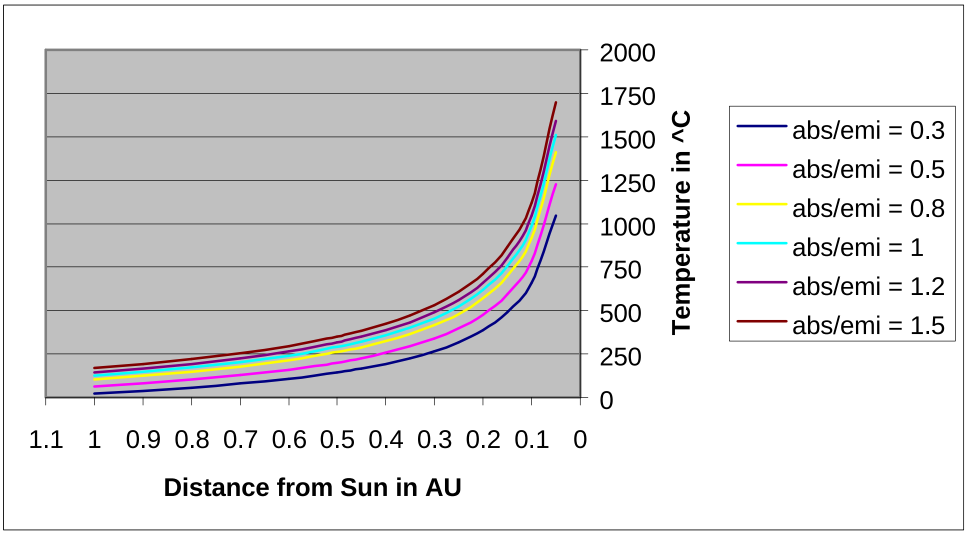


Figure 7-2: Temperature of a generic satellite surface vs the absorptivity/emissivity (abs/emi) ratio and as a function of solar constant (theta = 0°)

Thermal constraints on the HGA limit its use to distances >0.5AU from the Sun. The spacecraft is 3-axis stabilised always sun pointed (X-axis) except during SEP firing. Its Sun pointing face is as small as possible to minimise solar external fluxes and to use remaining satellite surfaces walls as radiators. With that assumption, only one thermal shield made of 5 titanium foils plus 20 layers Kapton/Dracon net shall be used to protect spacecraft bus and spacecraft Z faces during the various mission design cases:

- Cruise: SEP firing at 0.33 AU with +/- 10deg off X-axis pointing
- Observation: minimum Sun/Spacecraft distance 0.21 AU

Furthermore, thermal shielding is also designed to keep HGA and solar arrays mechanisms within standard temperature limits (temperatures experienced at 1 AU) to increase reliability.

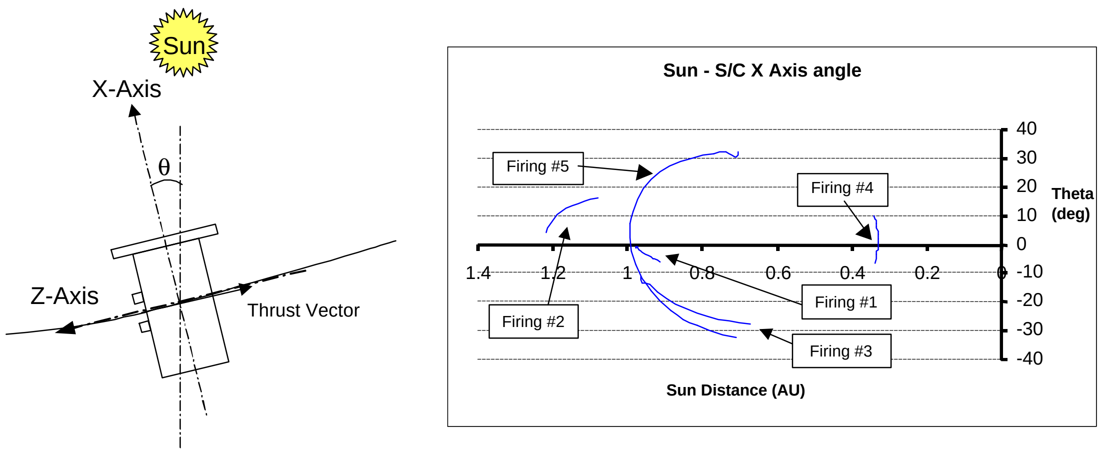


Table 7-1 –SEP Firing Philosophy

7.2.2 Power Supply

Two sets of solar arrays are required, one for the cruise, and one for the orbit phases of the mission.

The cruise solar array is required primarily for the electrical propulsion, but due to its impact on the pointing accuracy required by the instruments it will be jettisoned at the end of the cruise phase. Furthermore, it would be more complex to design and operate a solar array compatible with both phases due to the thermal environment close to the Sun. The cruise solar array will take advantage of a standard design (for GEO Missions).

The orbit solar array is different to that used in the Cruise phase in order to increase the upper temperature limit.

Extensive work was performed to define the strategy for Sun incidence variation on solar cells to prevent maximum temperatures being exceeded during both Cruise and Nominal+Extended mission phases. The solar aspect is controlled to reduce the thermal effect on the solar cells whilst generating sufficient power to support operations.

The figure below shows how Sun incidence angle can reduce the temperature of the solar arrays. See Solar Array chapter for more information.

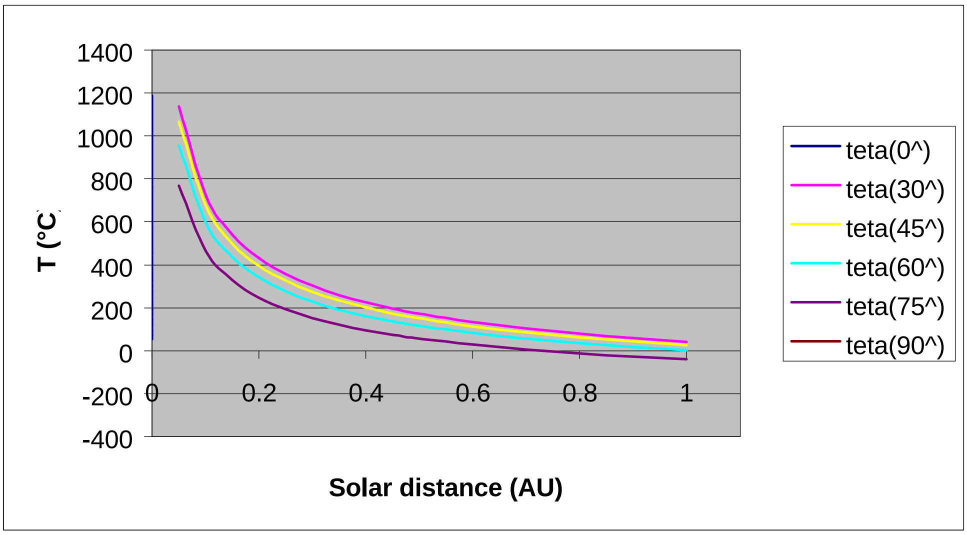


Figure 7-3: Solar Array Temperature vs Distance from Sun and Solar Angle Incidence.

7.2.3 Data Collection/Dumping

Trade-offs have been made to identify the data transmission strategy to Earth, considering the variable transmission duration and distance from Earth with each orbit.

Three sets of observation periods (ten days each) are considered as baseline, with an observation strategy tailored for each orbit:

1. Maximum southern latitude
2. Maximum northern latitude
3. Perihelion.

During High rate data acquisition (75 Kbps, in Nominal Observation Mode), see figure below, on-board data storage (240 Gb memory) is foreseen. Data dumping to ground shall occur when the Sun/Spacecraft distance is ≥ 0.5 AU and when no high rate observations are being performed.

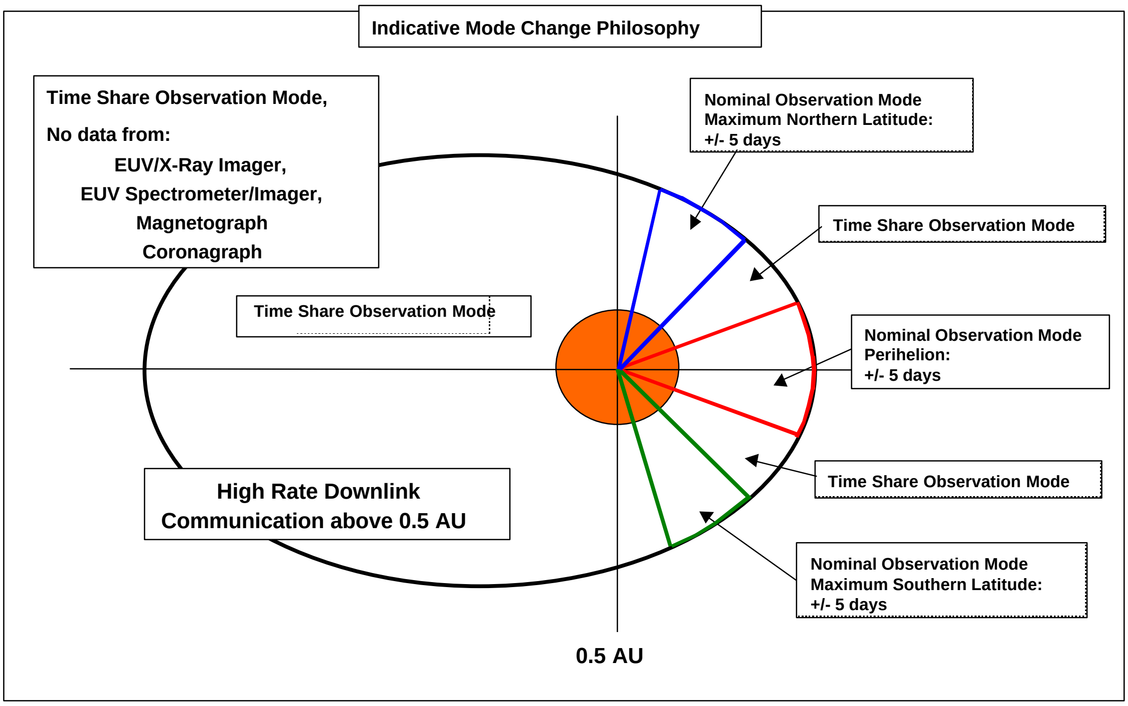


Figure 7-4: Modes of Operation during Nominal and Extended phases.

7.2.4 Radiation

An evaluation of environment radiation dosage over three mission phases was made. Results indicate that a nominally shielded (4mm Al.) silicon component is expected to receive a dose of 48Krads over the entire mission. The total ionising dose is within current engineering standards. Detailed results of this evaluation are reported in the Appendix.

7.2.5 Solar Sail – Initial Assessment

As a further trade-off, an initial assessment of Solar Sail use applied to the Solar Orbiter was made. Results are detailed in report Ref. 1.

7.2.6 Definition of Spacecraft Modes of Operation

Number	Mode Name	Definition	Acronym
1	Launch Mode	*Onboard launcher* - this mode is used until deployment of the solar array. All sub-systems are off, **except essential equipment (e.g.RX, CDMU, RTU)**. An automatic switch is used at separation to activate the equipment start-up sequence (tbc, launcher dependent). Satellite capable of receiving and executing Telecommands (e.g. SA deployment).	LM
2	Initialisation Mode	***Initial Deployment and Attitude acquisition:*** SA deployed and operational Attitude acquisition with SUN pointing Service Module Commissioning - all s/s in nominal working status Payload Module Commissioning - all units in nominal working status (tbc, deployment, etc) TT&C by LGA. **Contingency situation possible**	IM
3	Cruise Mode	***Mode for transfer from Earth to Solar Orbit:*** No Science operation during this mode. Instrument occasional tests, when SEP not active. (no additional power demand) The spacecraft is kept SUN pointing. Except during SEP Thrusting The SEP propulsion system is in operation during 5 firings. TT&C by LGA. Ejection of SEP SA (tbc), after s/c in final elliptical operational orbit. **Contingency Situation possible**	CM
4	Nominal Observation Mode	***Nominal Solar Observation:*** The spacecraft is kept SUN pointing. Accuracy determined by instruments Collection of science data from all instruments. 100% use. Storage of collected science data. **Contingency Situation possible**	NOM
5	Time Share Observation Mod	***Nominal Solar Observation:*** The spacecraft is kept SUN pointing. Accuracy determined by instruments No data from EUV, UVS, MAG and Coronagraph Science telemetry data sent back to Earth by HGA. **Contingency Situation possible**	TSM
6	Safe Mode	***Hibernation and Failure Recovery mode:*** The spacecraft is kept SUN pointing. Accuracy determined by power system. Instruments are put on standby or switched off. Non-essential functions are halted. TM/TC access to DHS is guaranteed to enable failure detection and reconfiguration. TT&C by LGA. Failure detection and recovery are executed by the ground. **Contingency Situation possible**	SM

Contingency Situation	***No link (in principle TC) between ground station and spacecraft.*** The s/c shall remain in current operating mode autonomously for at least 72 hours.(decision making mechanism, tbd) TC detection protocol to be defined together with ESOC. (e.g. elapsed time between TCs.) Science data should be stored during this mode, without loss of information.

Table 7-3: System Modes of Operation

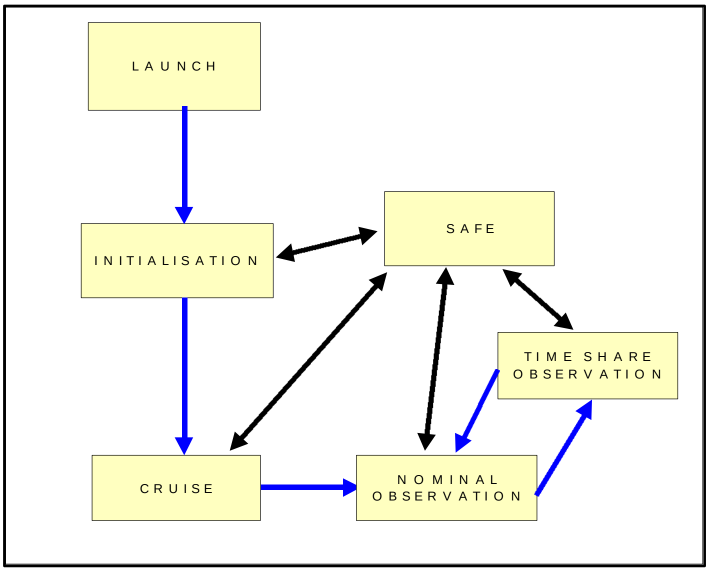


Table 7-2: Changes in System Modes of Operation

7.3 System Budgets

7.3.1 Mass Budgets

The mass identified in the system budget is based on the specified values of the individual units and subsystems. Depending on the maturity status of the items, contingency is applied on unit/item level. For each piece of equipment a mass margin was applied in relation to its level of development:

- 5% Off-the-Shelf Items
- 10% Items to be modified
- 20% Items to be developed

A System level margin of 20% was placed on the spacecraft dry mass (dry mass including sub-system margins).

The Soyuz-ST Fregat launcher allows for:

- a 1560 kg (including adapter mass) spacecraft from Kourou
- a 1310 kg (including adapter mass) spacecraft from Baikonur

The design mass with margin (1510 kg) is below the figure given for Kourou launch site, giving an additional margin on mass of 50 kg.

Mass (including margin)	
2. Thermal Control	
4. Pyrotechnics	
5. Communications	
6. Data Handling	
7. AOCS (inc RCS)	
9. Power (inc Solar Arrays)	
11. Payload Allocation	
System Margin (20% on Dry Mass)	
Spacecraft Dry Mass	**1160 kg**
Propellant Mass:	
Main Burns	284 kg
RCS	16 kg
Adapter	50 kg
Total Launch Mass	**1510 kg**
Launcher Capability: Kourou	**1560 kg**
Launcher Capability: Baikonur	**1310 kg**

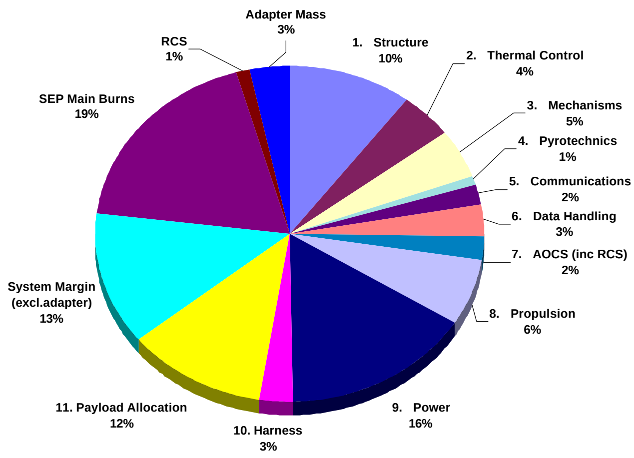


Table 7-4: Solar Orbiter System Mass Budget

7.3.2 Power Budget

Seven operational modes have been considering as dimensioning for the design of the power subsystem. The corresponding S/C power demand is given in the following table.

Mode		Instruments	Thermal	AOCS	Comms	Propulsion	OBDH	Power	Pyro	Mech	Harness	TOTAL CONSUMPTION	
Launch Mode		0.00	0.00	68.40	10.00	20.00	40.29	32.4	0.00	20.00	3.17	191.06	MAX
		0.00	0.00	48.20	10.00	10.00	30.50	22.2	0.00	10.00	2.17	130.87	MIN
Initialisation Mode		0.00	100.00	68.40	59.00	20.00	40.29	196.5	0.00	20.00	6.15	504.15	MAX
		0.00	100.00	48.20	10.00	10.00	30.50	189.7	0.00	10.00	4.17	398.35	MIN
Cruise Mode		0.00	60.00	68.40	59.00	6516.13	40.29	645.0	0.00	20.00	135.28	7544.12	MAX, power & Thermal S/S sizing
	Firing1 @ 0.99AU	0.00	60.00	68.40	59.00	4333.00	40.29	453.7	0.00	20.00	91.61	5125.98	Cruise Array sizing case 1
	Firing2 @ 0.96 AU	0.00	60.00	68.40	59.00	4709.00	40.29	469.2	0.00	20.00	99.13	5524.99	Cruise Array sizing case 2
	Firing3 @ 1.21 AU	0.00	60.00	68.40	59.00	3043.00	40.29	400.5	0.00	20.00	65.81	3757.03	Cruise Array sizing case 3
	Firing4 @ 0.98 AU	0.00	60.00	68.40	59.00	4354.00	40.29	454.5	0.00	20.00	92.03	5148.26	Cruise Array sizing case 4
		0.00	100.00	48.20	10.00	10.00	30.50	339.7	0.00	10.00	4.17	552.53	MIN
Nominal Observation Mode		127.00	60.00	68.40	59.00	20.00	40.29	52.3	0.00	20.00	7.89	454.91	MAX
		50.00	100.00	48.20	10.00	10.00	30.50	34.3	0.00	10.00	5.17	298.18	MIN
Time Share Mode		127.00	60.00	68.40	61.00	20.00	40.29	52.6	0.00	20.00	7.93	457.22	MAX
		50.00	100.00	48.20	10.00	10.00	30.50	34.3	0.00	10.00	5.17	298.18	MIN
Safe Mode		0.00	100.00	68.40	59.00	20.00	40.29	40.8	0.00	20.00	6.15	348.48	MAX
		0.00	100.00	48.20	10.00	10.00	30.50	27.7	0.00	10.00	4.17	236.37	MIN

Table 7-5: System Power Budget

7.3.3 Link Budget

The communication link calculations show that the required telemetry and telecommand links can be established.

Cruise Phase	Uplink	X - Band	34m Perth ⇒ LGA
	Downlink	X - Band	34m Perth ⇒ LGA
Nominal + Extended phases	Uplink	X - Band	34m Perth ⇒ LGA
	Downlink	X/KA - Band	34m Perth ⇒ LGA/HGA

Table 7-6: Communication Link Summary

The figures below show LGA max telecommand data rate vs S/C-Earth distance and HGA max telemetry data rate vs S/C-Earth distance respectively.

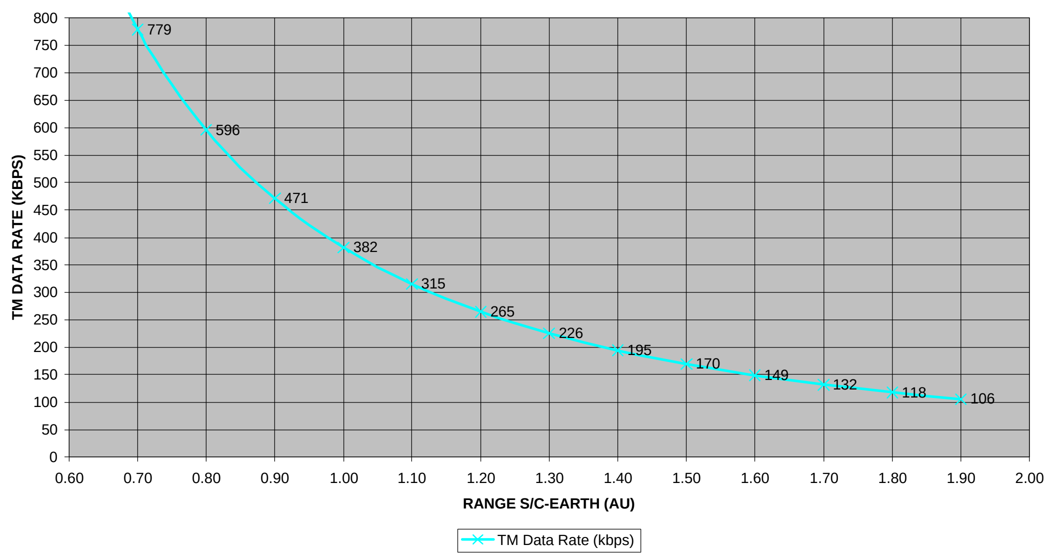


Figure 7-5: TM (Ka-Band) Data Rate against S/C-Earth Distance via HGA

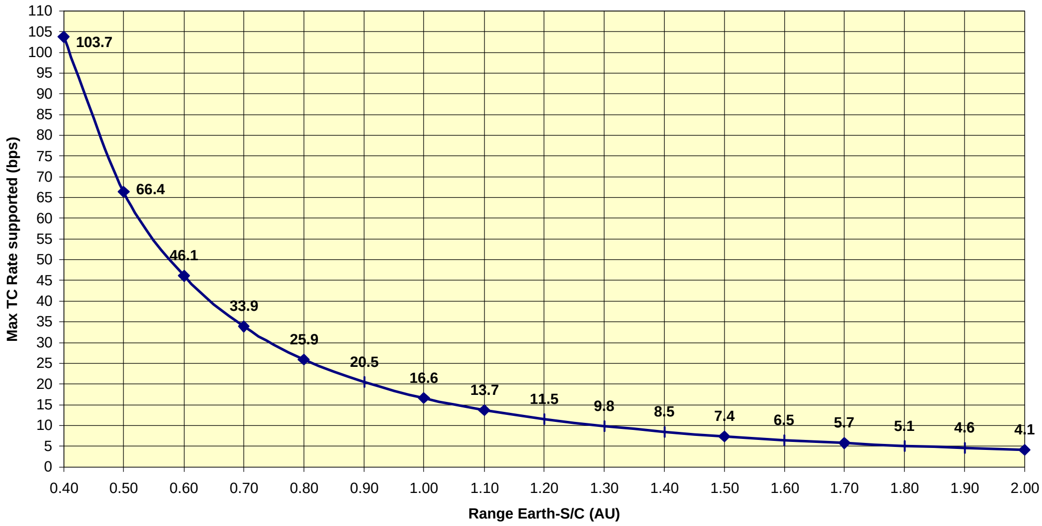


Table 7-7: TC (X-Band) Data Rate against S/C-Earth Distance via LGAs (Clear Sky Conditions)

S

System Equipment List

SOLO
1.33
01-Oct-99

					Mass with Margin
Total Spacecraft					
1. Structure		**131.36**	**19.21**	**25.23**	**156.59**
Adaptor ring	1	6.93	5.00	0.35	7.28
SVMCone	1	15.54	20.00	3.11	18.65
SVM+Y panel	1	6.02	20.00	1.20	7.22
SVM-Y panel	1	6.24	20.00	1.25	7.49
SVM+Z panel	1	5.16	20.00	1.03	6.19
SVM-Z panel	1	5.24	20.00	1.05	6.29
SVM X-Y shear wall					
SVM X-Z shear wall					
2. Thermal Control					
Optical Solar Reflector	-	2.85	5.00	0.14	2.99
Louvers	-	23.69	5.00	1.18	24.87
Other Thermal Hardware	-	14.50	5.00	0.73	15.23
Heat Pipes	6	5.25	10.00	0.53	5.78
High Temperature MLI	-	8.63	5.00	0.43	9.06
Doublers	-	5.60	5.00	0.28	5.88
3. Mechanisms		**67.30**	**10.00**	**6.73**	**74.03**
Cruiser Solar Arrays Mechanisms	NA	18.00	10.00	1.80	19.80
Orbiter Solar Array Mechanisms	NA	18.80	10.00	1.88	20.68
HGA Mechanisms, including Mast, APM, HRM.	NA	22.00	10.00	2.20	24.20
Cruiser Solar Arrays Mechanisms Electronics	1	3.00	10.00	0.30	3.30
Orbiter Solar Array Mechanisms Electronics	1	1.50	10.00	0.15	1.65
HGA Mechanisms, including Mast, APM, HRM. Electronics	1	4.00	10.00	0.40	4.40
4. Pyrotechnics		**9.70**	**5.00**	**0.49**	**10.19**
Cruise Solar Arrays HRM&JM	18	3.60	5.00	0.18	3.78
Orbit Solar Array HRM	6	1.20	5.00	0.06	1.26
HGA HRM	3	0.60	5.00	0.03	0.63
Deployable LGA HRM	0	0.00	5.00	0.00	0.00
Propulsion valves incl in propulsion mass	0	0.00	5.00	0.00	0.00
Pyro Control Unit	1	1.72	5.00	0.09	1.81
Pyro Harness and connectors	1	2.43	5.00	0.12	2.55
Safe Arm Connector	1	0.15	5.00	0.01	0.16
5. Communications		**28.60**	**6.26**	**1.79**	**30.39**
X/X-KA DST TRANSPONDER	2	7.00	5.00	0.35	7.35
KA BAND TWTA 20W	2	3.60	10.00	0.36	3.96
RF DISTRIBUTION UNIT	1	2.40	5.00	0.12	2.52
High Gain Antenna	1	10.00	5.00	0.50	10.50
Low Gain Antenna	4	2.00	5.00	0.10	2.10
X BAND POWER AMPLIFIER	2	3.60	10.00	0.36	3.96
6. Data Handling System		**41.70**	**9.26**	**3.86**	**45.57**
DHS	1	10.00	0.00	0.00	10.00
Mass Memory	2	13.90	20.00	2.78	16.69
RTU2	2	10.80	10.00	1.08	11.88
AOCS I/F	1	7.00	0.00	0.00	7.00

System Equipment List

SOLO
1.33
01-Oct-99

					Mass with Margin
7. AOCS		**34.11**	**7.19**	**2.45**	**36.57**
Star Tracker Head + Baffle	2	1.26	10.00	0.13	1.39
Star Tracker Electronics	2	2.34	10.00	0.23	2.57
Inertial Reference Unit	2	8.20	10.00	0.82	9.02
Sun Acquisition Sensors	3	0.69	5.00	0.03	0.72
Reaction Wheel	4	10.20	5.00	0.51	10.71
Reaction Wheel Electronics	1	3.12	10.00	0.31	3.43
Hydrazine Thrusters	12	4.20	5.00	0.21	4.41
Tanks and Prop. Feed	1	3.79	5.00	0.19	3.98
Pipes and Harness	1	0.31	5.00	0.02	0.33
8. Propulsion		**88.98**	**6.52**	**5.80**	**94.78**
Thrusters	4	25.00	5.00	1.25	26.25
PPU	2	27.00	10.00	2.70	29.70
Tanks and Prop. Feed	1	33.56	5.00	1.68	35.24
Pipes and Harness	1	3.42	5.00	0.17	3.59
9. Power		**219.84**	**9.40**	**20.66**	**240.50**
Power Conditioning Unit	1	43.61	5.00	2.18	45.79
Power Distribution Unit	1	8.60	20.00	1.72	10.32
Cruiser Solar Array	2	103.22	10.00	10.32	113.54
Orbiter Solar Array	2	56.41	10.00	5.64	62.05
Battery	2	8.00	10.00	0.80	8.80
10.Harness		**33.86**	**20.00**	**6.77**	**40.63**
Power Harness	-	23.86	20.00	4.77	28.63
Data Harness	-	10.00	20.00	2.00	12.00
11. Instruments (Payload)		**145.00**	**20.00**	**29.00**	**174.00**
EUV/x-ray Imager	1	15.00	20.00	3.00	18.00
EUV Spectrometer/Imager	1	50.00	20.00	10.00	60.00
Plasma Wave Package	7	9.00	20.00	1.80	10.80
Solar wind Analyser	1	3.00	20.00	0.60	3.60
Particle Detector	2	2.00	20.00	0.40	2.40
Dust Detector	1	1.00	20.00	0.20	1.20
Magnotograph	2	40.00	20.00	8.00	48.00
Coronagraph	1	25.00	20.00	5.00	30.00
Service and Payload Module DRY MASS (without overall margin)		860.97			967.04
				System Margin	193.41
				SYSTEM TOTAL DRY MASS	1160.45
				Propellant Mass:	
				Main Burns+Residuals	283.57
				RCS	15.50
				SYSTEM TOTAL WET MASS	**1459.53**
				Launcher Adapter	50.00
				TOTAL LAUNCH MASS	**1509.53**

Table 7-8: System Equipment List

8 Configuration

8.1 Requirements and Constraints

The major drivers for the overall configuration of Solar Orbiter can be summarised as follows:

- Payload: instrument sensor sun pointing (except for the magnetometer) with no obstacles in the fields of view. CCD radiators, assumed mounted on the sensor body, open to cold space. Instrument stable mounting and accessibility. Accommodation of the 2.8 m long EUV spectrometer/imager.
- Thermal: sun input reduction, radiating surface.
- Propulsion: SEP thruster line of action through the S/C centre of gravity during the whole cruise phase.
- Launcher: accommodation of the S/C in stowed position under the SOYUZ type S fairing and mechanical interface with FREGAT
- Power: Accommodation of solar array area in two sets: cruise SA and orbit SA.
- Communications: accommodation of the large HGA
- Others: equipment mounting area and particular equipment accommodations.

The S/C must provide accommodation to all the subsystems and ensure compatibility between them throughout the mission. Therefore each of the constraints above must be observed for every operational mode, sun distance range and sun-earth S/C attitude.

8.2 Spacecraft baseline design

Figure 8-1 to Figure 8-4 show the Solar Orbiter S/C overall configuration. The S/C body is a prism 3000(X) x 1600(Y) x 1200(Z) mm. The +X side, facing the sun (+/-30deg), is covered by a thermal shield shadowing the S/C body. In order to minimise the energy input from the sun the S/C body footprint area has been minimised.

The S/C is modular, Service Module (SVM) and Payload Module (PLM), with some overlapping between both.

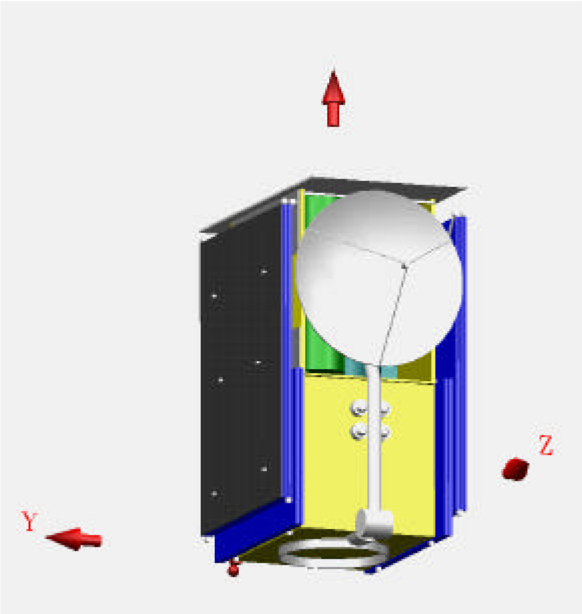
The +/-Y sides of the PLM accommodate the cruise solar array (2 wings, 3 panels per wing) and the top shield radiators. The optical instruments are right beneath the top shield (Figure 8-5), pointing +X, isostatically attached to the central cylinder and are open to cold space in the +\- Z sides. The instrument electronic boxes are mainly mounted on the bottom panel of the PLM.

The SVM accommodates the orbit solar panels and the SEP/equipment radiators on the +/-Y panels. The HGA and the thrusters are attached on the +Z panel. The propellant tank is at the Centre Of Gravity (COG) of the S/C, inside the central SVM cylinder mounted on a dedicated ring. This design can account for possible changes on the COG height through the S/C design. The equipment boxes are mounted internally on the SVM panels. The internal accommodation has been assessed in terms of available mounting area complemented with a check on selected particular accommodations (Figure 8-6).

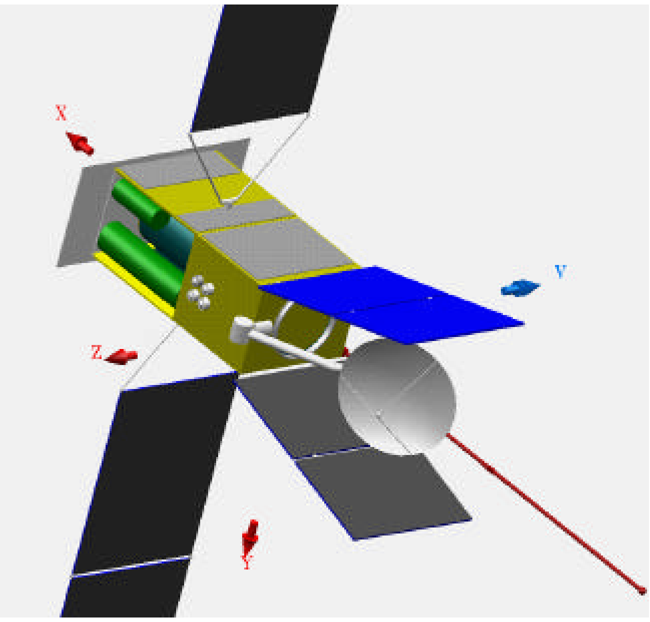
In order to cope with different attitudes and distances from the sun, both solar arrays incorporate 1 Degree of Freedom (DOF) driving mechanism. The cruise SA can be jettisoned. The HGA mast (2m long) is mounted on a 2 DOF mechanism and ensures coverage throughout the mission. Four configurations are considered (Figure 8-1):

- Launch: solar arrays, HGA and magnetometer stowed
- Cruise: Cruise solar array deployed 19 m tip to tip. Orbit solar array deployed 75 deg w.r.t. Y-axis. HGA protected from the sun behind the orbit solar array. SEP thrusters firing laterally in +Z direction.
- Orbit/Observation: Cruise solar array jettisoned. HGA protected behind the orbit solar array
- Orbit/Downlink: orbit solar array angle between 0 and 75 deg w.r.t Y-axis, HGA in operational position.

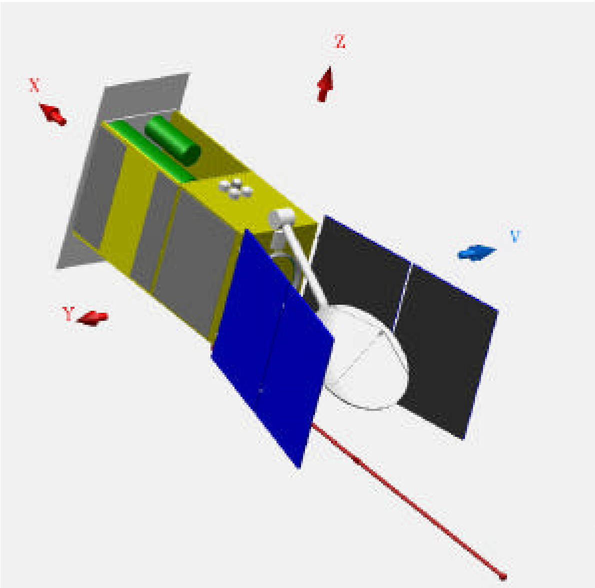
The Plasma Wave Package accommodation has not been studied due to a lack of instrument definition data.



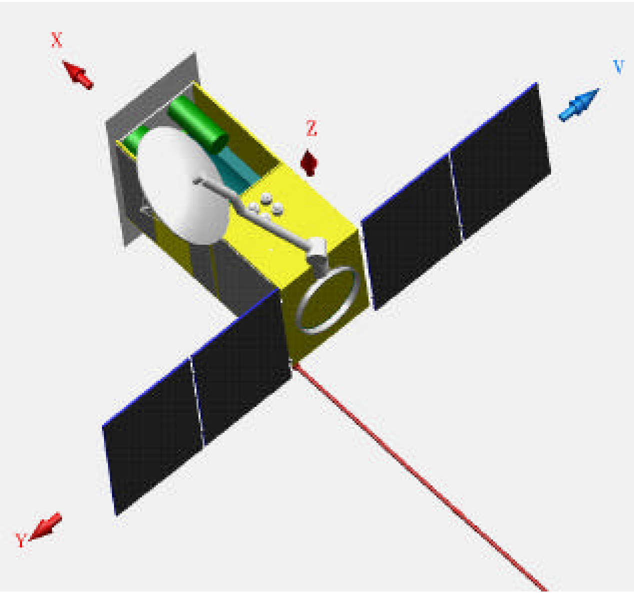
Launch



Cruise



Observation



Downlink

Figure 8-1: Solar Orbiter configurations

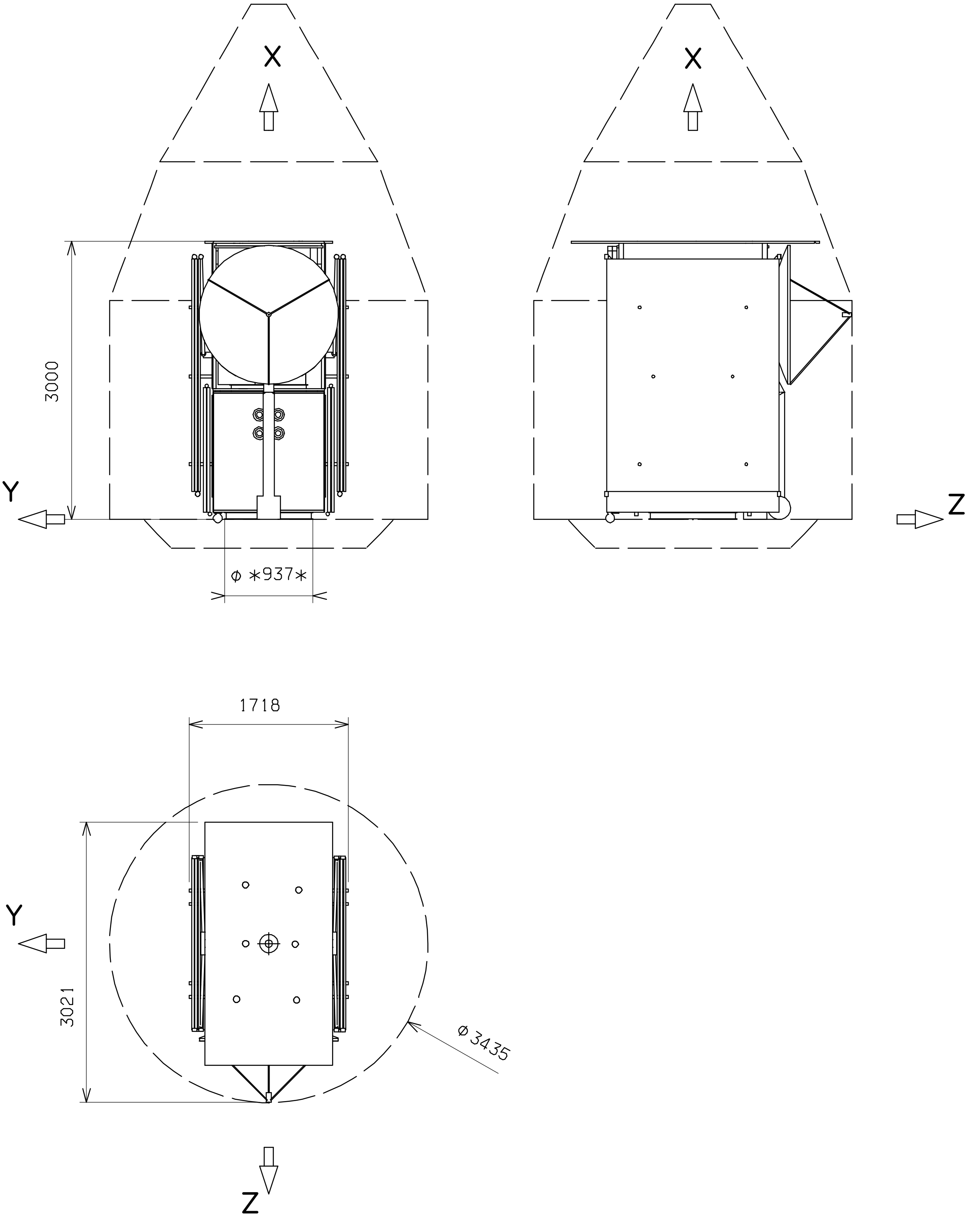


Figure 8-2: Solar Orbiter main dimensions. Launch configuration.

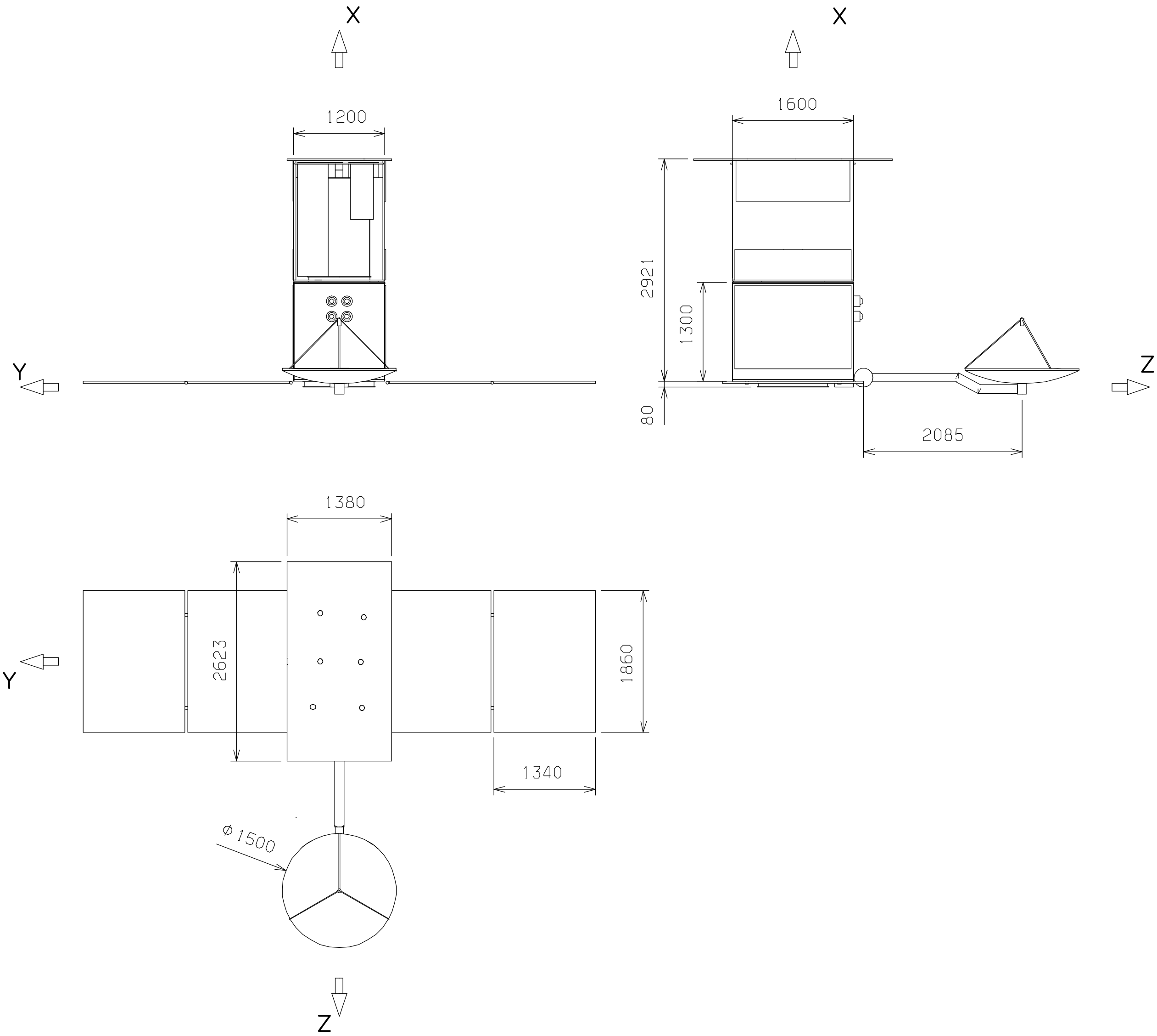


Figure 8-3: Solar Orbiter main dimensions. In-orbit configuration.

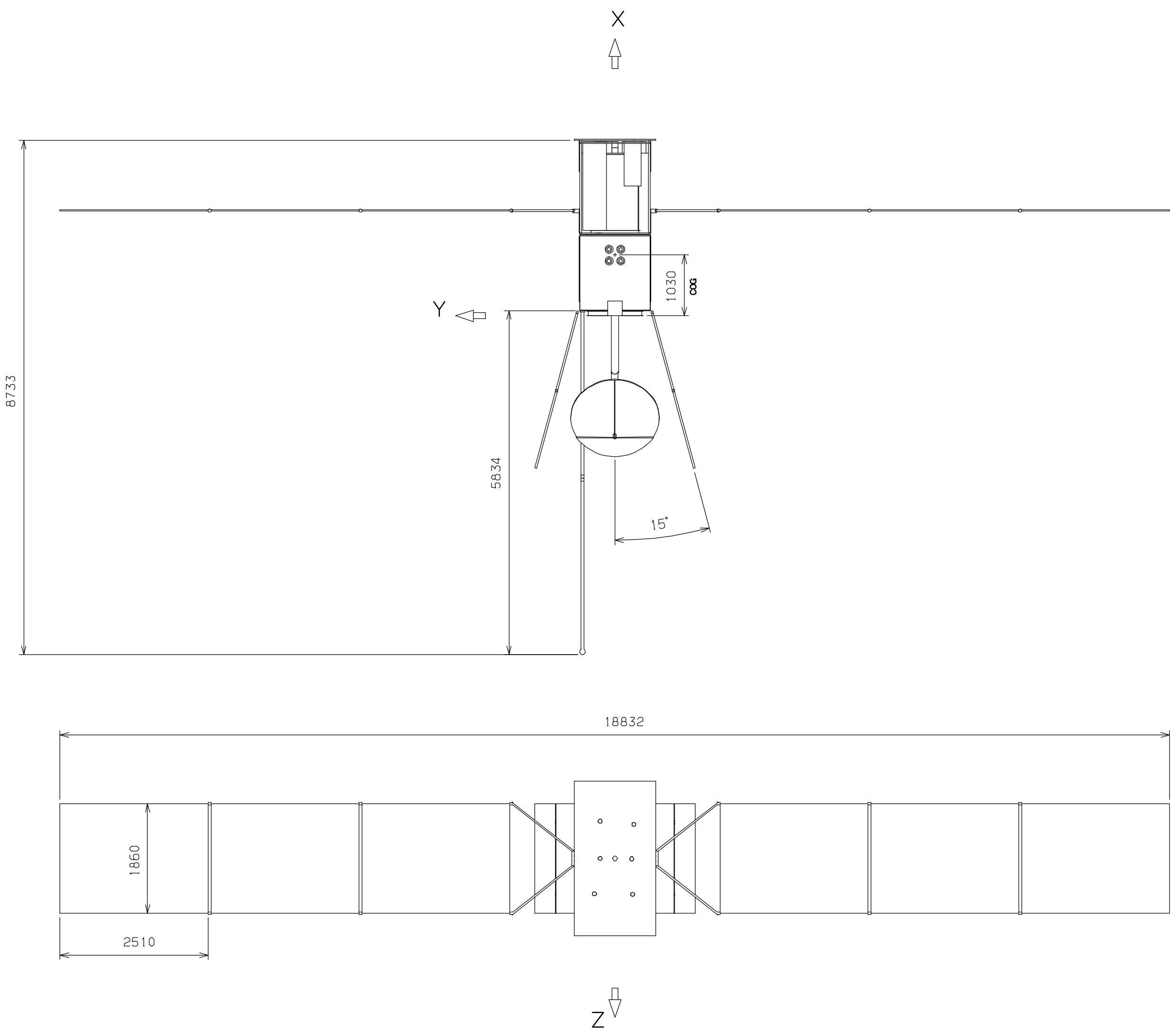


Figure 8-4: Solar Orbiter main dimensions. Cruise configuration.

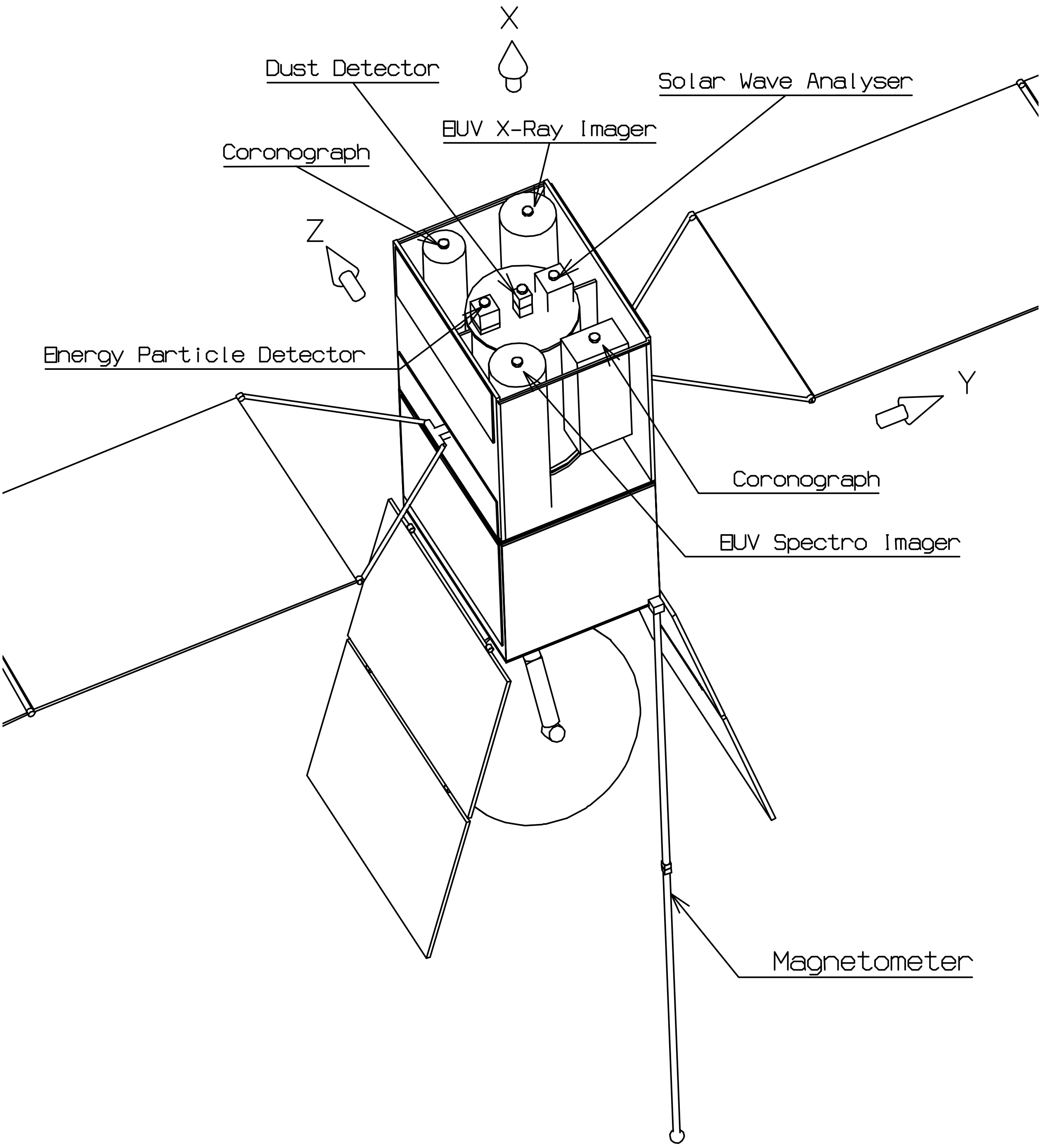


Figure 8-5: Solar Orbiter instruments accommodation.

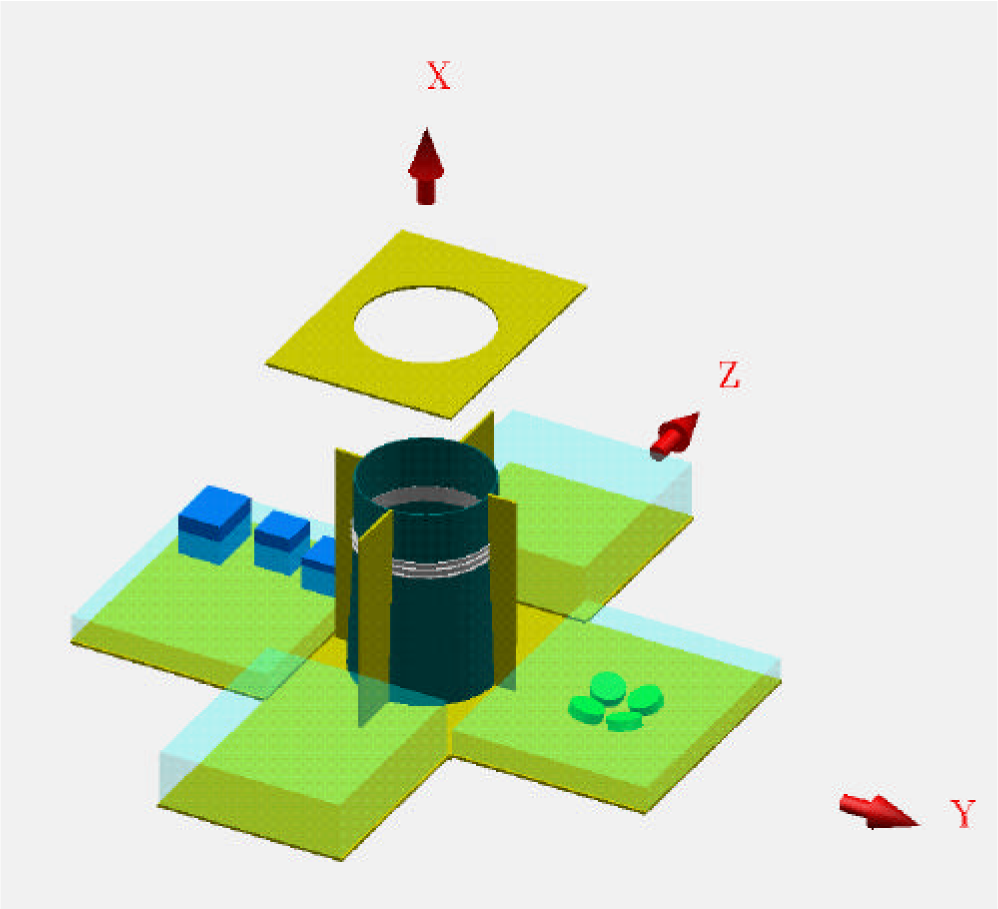


Figure 8-6: Solar Orbiter internal accommodation assessment. Available volumes and particular accommodations.

9 PROPULSION

9.1 Subsystem Requirements and Design Drivers

The main requirement for the propulsion subsystem is to enable the completion of Earth to solar orbit manoeuvres, as designed in the mission analysis. This translates into the total delta-V figure of 4770 m/s, with an additional 140 m/s for the Gravity Assist auxiliary manoeuvres. Such a high delta-V drives the selection of the thrusters towards a high I_{sp} system, in order not to exceed the launch mass limit of 1560 kg imposed by the selected launcher.

A second set of requirements is originated by the Attitude Control subsystem. The total angular momentum needed for the offloading of the reaction wheels is 15932 Nms, and a minimum thrust level of a few N is desired for the Rate Reduction and Safe modes.

As already stated in the previous sections of this report, the design of a spacecraft for a “Flexible” type of mission is constrained mainly in the selection of existing (“off-the-shelf”) technology: this is a major design driver for the propulsion subsystem too.

9.2 Design Assumptions and Trade-Offs

Electric propulsion is selected for the cruise phase of the mission, due to the high delta-V required.

Ion Thrusters and Stationary Plasma Thrusters are both considered as viable options, the higher I_{sp} of the first system leading to a smaller amount of propellant but to a higher power demand than the second. Since both the total mass and the power management are critical issues of the mission, a trade-off could be considered between the two thruster systems. In the frame of this study, however, the technology requirement has lead to the immediate selection of SPTs: a 200 mN SPT thruster is in fact being currently developed and is expected to be qualified before 2003.

All figures (i.e. mass, power consumption) used in the sizing of this system are based on the above mentioned development, and therefore include the expected improvements in terms of masses and efficiencies.

A second propulsion module is used as reaction control system. Hydrazine thrusters are baselined (see section 9.2.2). The propellant mass is obtained from the required angular momentum assuming an average arm length between thrusters of 0.5 m.

The effect of the radiation environment is accounted for through an increase in the power units’ mass margin.

Two main trade-offs have been considered during the study, related to the operation of the SPTs and the selection of the attitude control thrusters. Both are briefly summarised in the following sections.

A preliminary assessment of the possible use of solar sails as an alternative propulsion system has been performed during the study, see Systems chapter.

9.2.1 SPT firing strategy

In order to ease the thermal control of the spacecraft, a Sun-pointing attitude is required throughout the mission. The thruster placement on the +Z panel is a consequence of this requirement. During firing the spacecraft is positioned up to a maximum of 30° (Sun-S/C X-axis angle) off of Sun pointing. The following analysis has been made in order to assess the mass and power penalties involved by a change in firing strategy that would allow Sun pointing to be retained during the thrust phases.

Trade-Off Summary		Baseline *No Canting*	Option #1 *30 deg Canting*	Option #2 *10 deg Canting*
		thrust-pointing during all firings; thrusters position not relevant	Sun-pointing throughout all cruise, thrusters on +Z	Sun-pointing during firing #4 (min Sun distance), thrust-pointing during other firings; thrusters on +Z
Subsystem				
Thrust % provided by one thruster	**%**	50	100	100
Number of firing thrusters	**-**	4	2	2
Maximum cant angle	**deg**	0	32.7	9.96
Average cant angle	**deg**	0	27.94	8.53
Results				
Total Wet Mass	**kg**	390.1	477.9	443.6
Mass Delta	**kg**	0.0	87.8	53.6
Max Power	**W**	6,516.13	7,698.96	6,582.59

Table 9-1: SPT Firing Strategy Trade-Off

According to this strategy, the thrusters would be canted by the maximum offset angle mentioned above, and thrust modulation would be used in order to rotate the thrust vector and keep it tangential to the orbit while the spacecraft would remain Sun-pointing. The thrusters would have to be placed in a line, at the intersection of the orbit plane with the +Z panel; only two thrusters would fire at the same time (one “left” and one “right”), while the other two would be the redundant pair. Each thruster must be capable of providing the maximum (0.3 N) thrust when aligned with the thrust direction.

A reduced version of this strategy has been studied, where the Sun-pointing attitude is only kept during the fourth firing, the most critical from a thermal point of view (minimum Sun distance). In this case, the maximum angle between the Sun-S/C line and the S/C X-axis would only be of 10°.

For the purpose of the study, since the above mentioned angle is a function of time, its maximum and time-averaged values were used for the sizing of propellant and power.

As table below shows, option 1 is affected by too heavy mass and power penalties to be considered. The mass increase of the Sun shield that protects the spacecraft from a ±10° Sun incidence angle is much lower than 65.7 kg, so that option 2 is also not worth considering.

It should be noted that the different firing strategy suggested (requiring 300 mN per thruster and some enhancements to the modulation capabilities of the foreseen SPT technology) do not fulfil the technology readiness requirement.

9.2.2 AOC thrusters

Four different propulsion systems were initially considered in this study as RCS actuators: Cold Gas, Hydrazine, Resistojet and FEEP thrusters.

The Cold Gas system was immediately seen to be too heavy, due to the very low specific impulse and high tank and fuel mass fraction.

Resistojets were considered attractive because, if used with Xenon as a propellant, they could use the SPT tank; the mass saving obtained in this case would however be offset by the relatively high power required. Since both this system and the Hydrazine thrusters have roughly the same specific impulse, the latter were considered a better option.

A trade-off study was then carried out with the FEEP thrusters as an alternative option, in order to investigate the extent of the propellant savings enabled by their very high I_{sp}.

Trade-Off Summary	BASELINE Hydrazine + RW	OPTION #1 Hydrazine with FEEPS		OPTION #3 FEEP only
	Hydrazine thrusters used for RW offloading	Hydrazine thrusters used in Rate Reduction Mode, Safe Mode and SPT Mode only		FEEP thrusters used as only AOC actuators (without RW)
Subsystem				
Thruster Type	HYDRAZINE	HYDRAZINE	FEEP	FEEP
Thrust Level (per thruster)	5 N	5 N	0.75 mN	0.75 mN
Number of PPUs	0	0	2	4
Number of Thrusters	12	12	8	16
Total Impulse (Ns)	31,864.40	18,408.24	13,456.34	31,864.40
Results				
Dry Mass (kg, incl. margin)	8.4	7.2	12.2	20.8
Reaction Wheels Mass (kg)	13.5	13.5		0.0
Propellant Mass (kg)	13.6	7.9	0.2	0.6
Wet Mass (kg, incl. margin)	35.5	41.1		21.4
Input Power (W)	*20 (peak)*	*20 (peak)*	*95.2*	*95.2*

Table 9-2: Attitude Control Thrusters Trade-Off

The first option considers the use of FEEPs instead of Hydrazine thrusters for the RW offloading during those phases of the mission where a very low thrust is acceptable; in the second option all attitude control manoeuvres are performed with FEEPs.

The table above shows that the propellant savings achieved in the first option are not sufficient to compensate for the increase in dry mass, complexity and power demand of the whole system.

The second option is based on a completely different attitude control strategy: more thrusters are available (16 instead of 12) for a more accurate control of the spacecraft with a lower total mass, and the expected availability of four fully integrated clusters (of four thrusters each) would significantly ease their accommodation. However, the high power request and the low thrust (much lower than the desired value), together with the expected higher cost of this system have oriented the selection to the baselined hydrazine system.

9.3 Subsystem Baseline Design

The Solar Electric Propulsion (SEP) module is made up of four SPT thrusters, controlled by two Power Processing Units (PPU). One tank with the standard set of propellant management units feeds the thrusters (table below).

The thrusters are symmetrically placed around the spacecraft centre of mass, on the +Z panel (see configuration and 9.2.1); the required thrust in every phase of the mission is achieved by the contemporary firing of all thrusters. Thrust modulation, well within the technological capability of the SPT thrusters and electronics, allows the attitude control along the X and Y axes to be performed with the SPTs in order to save RCS propellant.

In case of failure of one thruster, the opposite one will be switched off and the remaining two operated at double thrust level: with this strategy the sizing thrust level is 150 mN, which falls within the 200 mN range provided by the on-going technological development.

The Plasma Thrusters will also be used for the Gravity Assist preparation manoeuvres during the Nominal and Extended mission phases: the power will be provided, in this case, by the orbit solar arrays.

Twelve Hydrazine thrusters are used as RCS actuators, arranged in six pairs of cold redundant thrusters (table below). One tank, operating in blowdown mode, supplies the required propellant.

MASS AND POWER BUDGETS			
Propulsion Modules		**SEP**	**AOC**
Thrusters	kg	26.3	4.4
PPUs	kg	29.7	0.0
Tanks and Propellant Feed	kg	35.2	4.0
Pipes and Harness	kg	3.6	0.3
Total Dry Mass	kg	***94.7***	***8.7***
Propellant	kg	**283.1**	**15.5**
Total Wet Mass	kg	***377.9***	***24.2***
Max Input Power	W	6,516.1	-

Table 9-3: Propulsion Subsystem Budgets

CONFIGURATION			
Propulsion Modules		**SEP**	**AOC**
Thruster Type	-	SPT	HYDRAZINE
Number of Thrusters	-	4	12
Thrusters Cant Angle	deg	0	0
Number of PPUs	-	2	0
Number of Tanks	-	1	1
Propellant Type	-	Xenon	Hydrazine

Table 9-4: Propulsion Subsystem Configuration

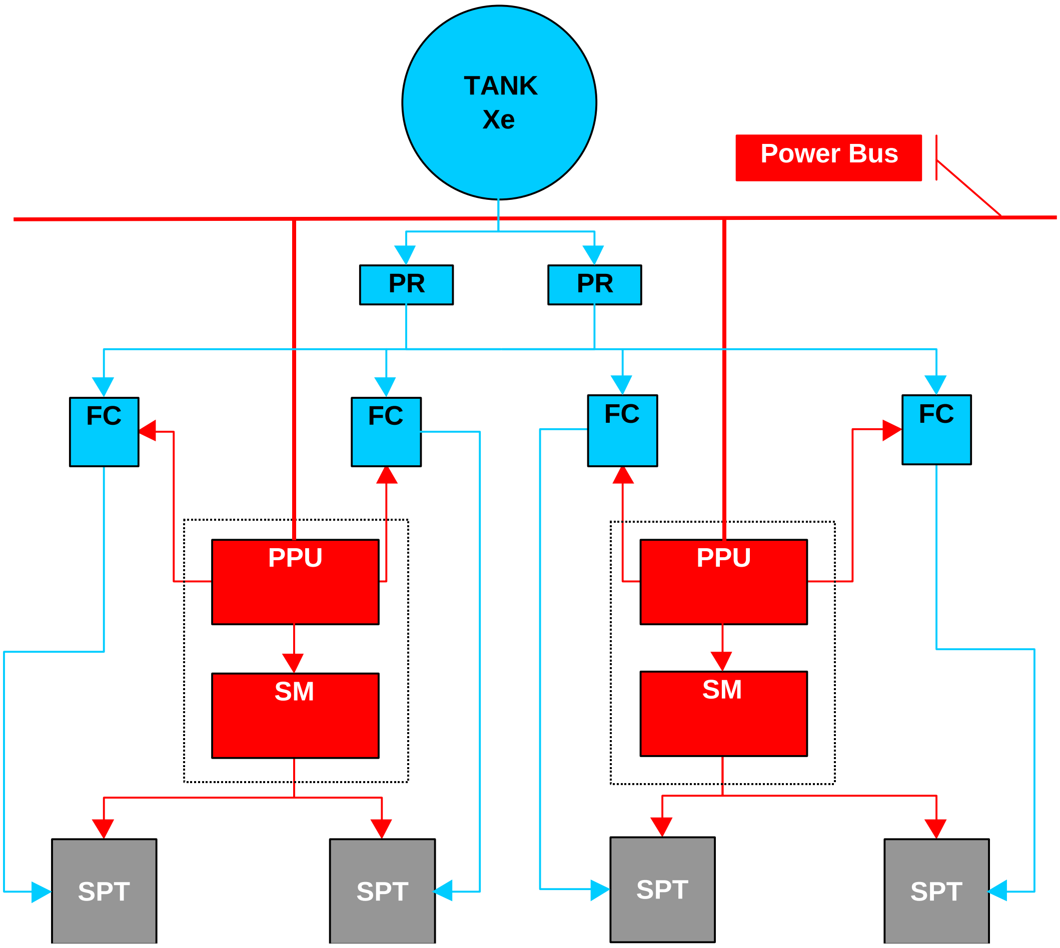


Table 9-5: Propulsion Subsystem Block Diagram

10 SOLAR ARRAYS

10.1 Cruise Solar Array Baseline Design

As a result of a detailed power budget analyses a mission profile has been defined outlining the different power needs at distinct points during the complex cruise phase. The cruise solar array area and capability has been optimised to provide the necessary power throughout the mission without requiring any operational constraints to the mission.

The cruise solar array design proposed for this mission is based on a standard solar generator type as it is currently used for GEO telecommunication missions.

The cruise solar array consists of two identical solar array wings comprising a yoke and 3 rigid panels each. The panels are standard honeycomb panels with CFRP face sheets with a dimension of 2510mm x 1860mm (Figure 10-1) resulting in a total panel area of 28m^2. The total solar array mass is 103 kg.

The end of life (EOL) power generated by this solar array is 6275 W at 1 AU in the maximum power point which corresponds to 5.2 kW at the operation voltage (50 volts bus). A detailed EOL power profile as function of the Sun distance is given in fig. 2.

The begin-of-life (BOL) installed power (AM0) is 8 kW at 25ºC corresponding to about 16,650 solar cells of 4 cm x 4 cm.

The solar cell assembly proposed for this mission is a dual junction GaAs based cell with a BOL efficiency of 22% at 25ºC covered by a 150 micron thick cover slide (e.g CMO). This approach is very conservative, since at the time of initiating a phase C/D for the Solar Orbiter mission, it is certain that triple junction solar cells will be available for large scale production. This would bring a reduction in required SA wing area of about 10%.

The worst case radiation fluence as predicted for the 7.2 year mission is 5.2 E14 1MeV equivalent electrons /cm^2. In order to cover any uncertainties for potential special effects due to the proximity to the Sun a worst case fluence of 1e15 1MeV el/cm^2 have been considered for the end-of-life prediction. As the accumulated fluence with time is not linear, for worst case considerations, the total fluence has been considered almost from the beginning of the mission. For this reason the SA power output graph as function of the distance to the Sun is only considering EOL conditions.

Since for most of the cruising time the actual generated power is far in excess of the required needs other typical losses have been discarded in the prediction. One of the power critical points is at 0.99 AU where 5.2 kW are needed. In this points the actual margin is 200 W with EOL radiation considered. Already at 0.02 AU less distance to Sun the power generated increases by ≈10% which is in the range of the typical additional losses considered for other missions. In case, that exactly during a thrust period close to Earth (full radiation dose considered), due to an exceptional failure, the power generation is less than the required thresholds, the propulsion system would have to operates slightly longer at a slightly lower thrust which is not seen to be a problem.

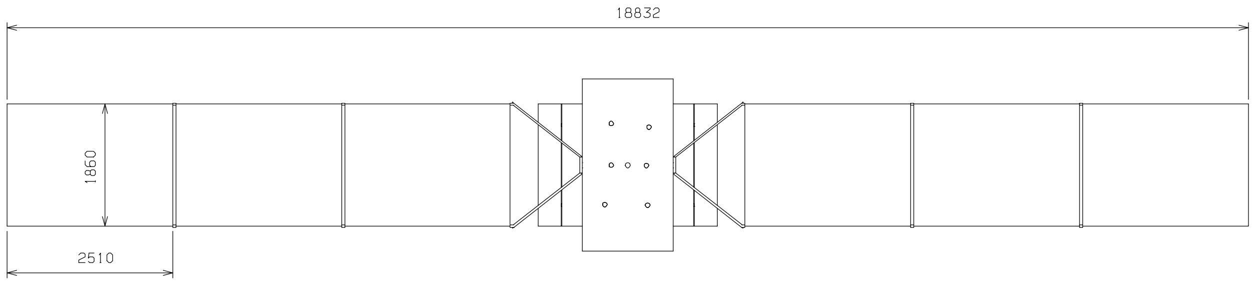


Figure 10-1: Cruise solar array (SA) configuration with SA wings normal to Sun (for range 1.22 AU –0.72 AU)

The solar array wings will be made rotatable (± 180 degrees) around one axis perpendicular to the trajectory plane in order to keep the operating solar array temperature safely below 130ºC (actual worst case cell temp. 106ºC, see fig. 3) in the astronomical range between 1.22 AU and 0.27 AU, and to provide the necessary power to the spacecraft. This overall strategy will be accomplished by initially keeping the SA wing perpendicular to the sun between 1.22 AU and about 0.72 AU. Below 0.7 AU the SA wings will be slowly rotated away from the Sun (proposed are small convenient steps with distance, see fig. 1.3). At 0.33 AU (closest firing of thrusters) the wings must be rotated to 75 degrees and between 0.33 AU and 0.27 AU (closest distance to Sun during cruise phase) the required tilt angle is 90 degrees (no power is generated).

For the Solar Orbiter mission this standard solar array requires to be equipped with thermal shields along the yoke and length sides of the panels (e.g. Titanium based plates attached by thermally isolating struts to the sun-facing panel length). They ensure that the panel edges and the yokes are not overheated when tilted. Some development effort will be needed to incorporate this modification into the standard design.

The system engineering type of thermal analysis showed that it is very promising that due to the relative homogeneous/flat surface of the electrical network no optical surface reflectors (OSR) or special cover glasses on the solar cells are required when the array is tilted to high rotation angles. However, in case a detailed analysis shows that the inter cell gaps are critical and result in unacceptable high hot spots, a solution similar to the design presented for the orbit array could be implemented. In any case, as soon as a specific design has been defined in detail for the cruise array, it is essential that a detailed thermal analysis is being performed and verified by test on representative samples.

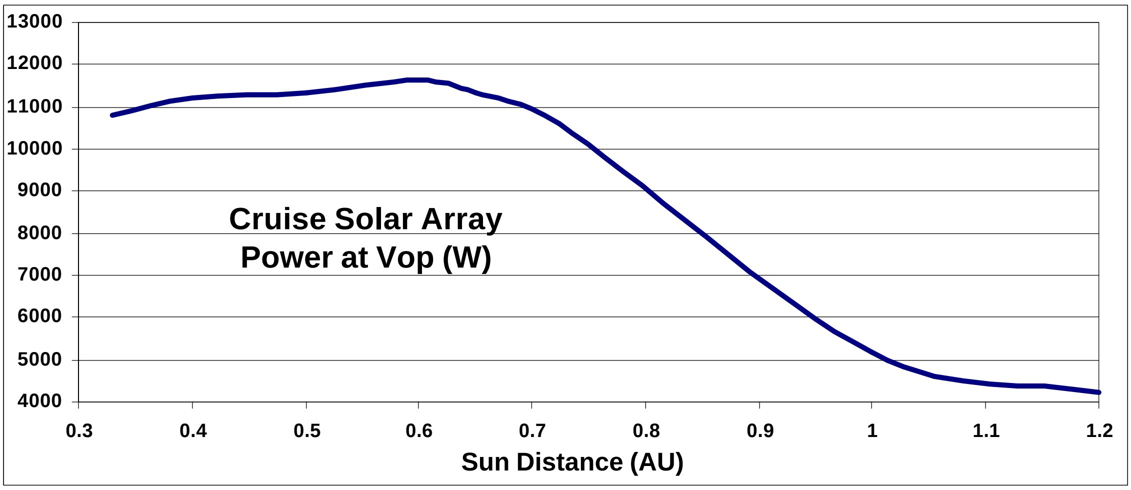


Figure 10-2: Cruise solar array power profile between 0.33 AU and 1.2 AU (EOL radiation fluence)

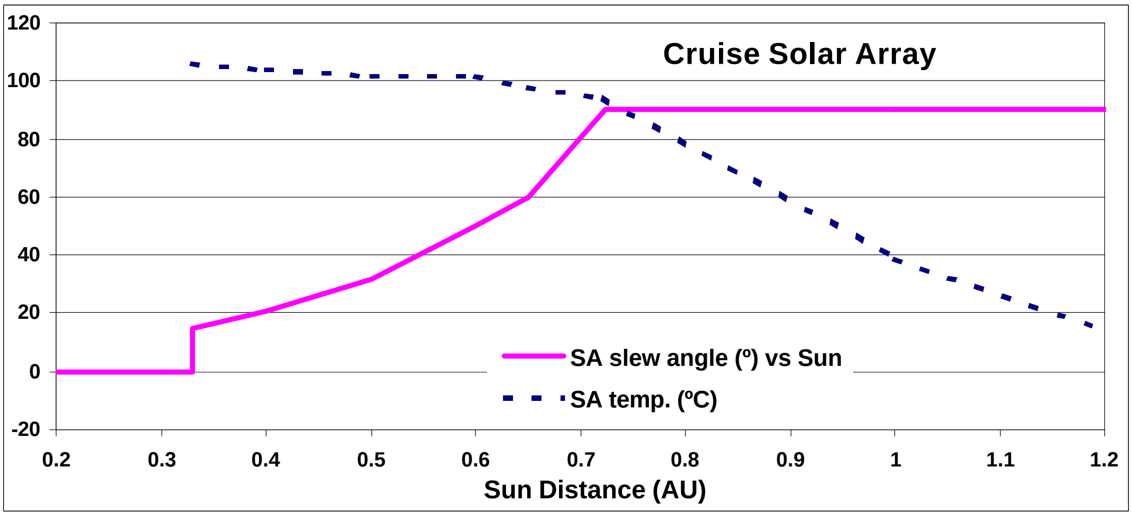


Figure 10-3: Cruise solar array temperature profile and Sun incidence angle on the SA between 0.33 AU and 1.2 AU

At larger Sun incidence angles, due to refraction and reflections, the power generation does not exactly follow the cosine law. This behaviour has been considered in the power and thermal analyses. The deviation from the cosine law is shown in Figure 10-4.

For the rotation of the wings, solar array drives (SAD) are used (one for each wing). It is a simplified version requiring no slip rings. A cable coil is sufficient to ensure the rotation of ± 180 degrees. The SAD will be shielded against direct sun illumination by a sunshield, which will require some development activity.

After completion of the cruise phase the cruise solar array wings jettisoned along with the solar array drive.

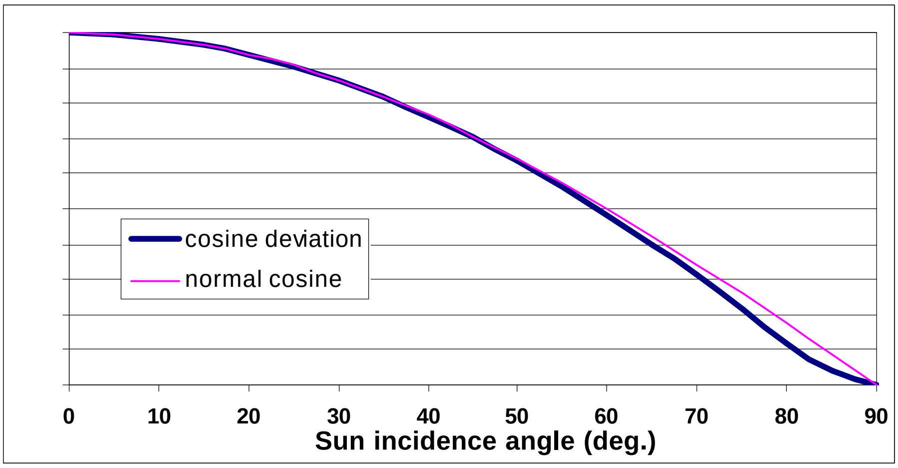


Figure 10-4: Cosine deviation at high Sun incidence angles relevant for power generation and thermal analysis

10.2 Baseline Design Description for the Orbit Solar Array

The orbit solar array has to survive the extreme environment between 0.21 AU and 1.22 AU and to provide the required power to the spacecraft in the range between 0.21 AU and 0.89 AU.

The orbit solar array consists of two identical SA wings comprising 2 rigid panels each. The panels are directly attached via rotatable hinges to the spacecraft side wall. These hinges are adjustable between 15º and 90º Sun incidence angles and have to follow the profile as given in Figure 10-8.

The panels are made of a honeycomb core with aluminium face sheets, which are required to reduce the thermal gradient within the panels when illuminated. All panels are identical and have a dimension of 1340mm x 1860mm (Figure 10-5).

The solar cell assembly proposed for this mission is a triple junction GaAs based solar cell with a BOL efficiency >24% at 25ºC covered by a 150 micron thick cover slide (e.g CMO).

The area of the orbit SA is10m^2. The total SA mass is 56.4 kg. The result from various system trades defined the operational range of the orbit solar array. During launch the orbit SA is stowed between cruise array and the spacecraft side wall. After deployment of the cruise SA both orbit SA wings are deployed and fixed at 15º Sun incidence angle and remain fixed until cruise array jettison (orbit array is fully protected against shadows). Just prior cruise array jettison orbit SA wings will be rotated to a Sun incidence angle required for this particular point (Figure 10-8). For the remainder of the mission the orbit SA wings have to be positioned as defined in Figure 10-8.

The orbit array has two major design drivers. At 0.21 AU the design must be capable to thermally survive the environment and the most distant point in the observation phase (0.89 AU) defines the area of the array needed to provide the 500 W to the spacecraft.

Sun area and capability has been optimised to provide the necessary power throughout the mission without putting any operational constraints on the observation plan.

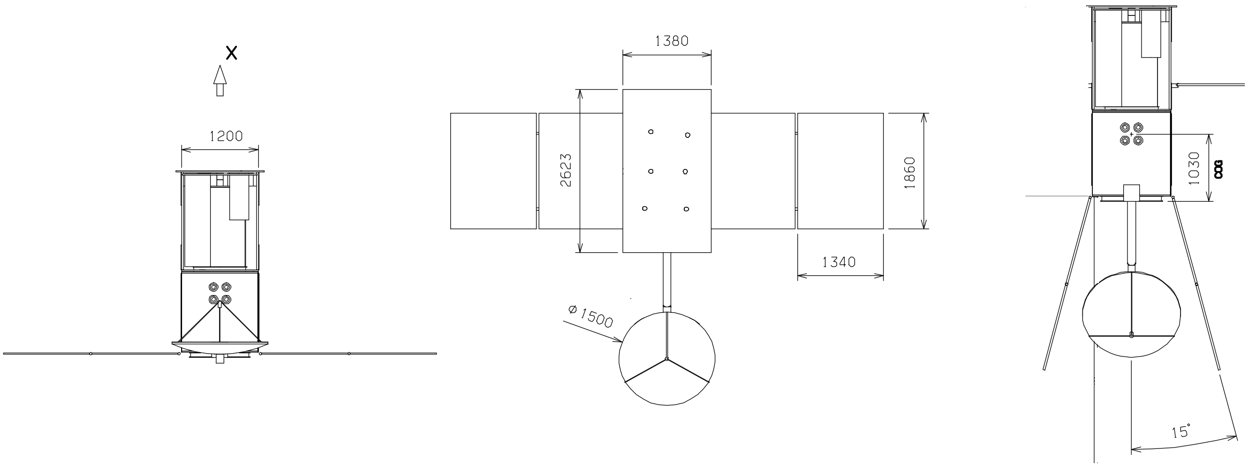


Figure 10-5: Orbit SA configuration with SA wings normal to Sun for range 0.50 AU –0.89 AU (left and centre) and at 15 degrees Sun incidence angle required at 0.21 AU (right)

In order to ensure that the solar array is never operated at temperature above its critical limit, the solar array wings will be equipped with second surface mirrors (SSM's) bonded between rows of triple junction solar cells. In the baseline a ratio 84% SSM’s and 16% solar cells are foreseen. The solar cell size shall be relatively small (e.g. 2 x 4 cm) to further reduce the temperature gradient between the SSM and cell area. Under worst case condition the solar cell assemblies reach 146ºC in the operational mode and 158ºC in the non operational mode (string shunted by S^3R). The basic concept of mounting SSM’s and solar cells next to each other has been successfully applied on the ISO solar array (

Figure 10-9, ratio ca. 65% cells, 35% SSM’s).

At the hinge lines the panels are equipped with thermal shields (e.g. Titanium based plates attached by thermally isolating struts). They ensure that the panel edges and hinges are not overheated when tilted.

Since the SSM’s and solar cell assemblies have different heights, if no precautions are taken, there will be local unacceptably high hot spots at the SCA edges (direct plus reflected light) under small incident angles (i.e. 15º). To counteract this problem the following preliminary design is proposed/recommended:

- reduced gap between SSM’s and cells as small as possible
- reduce the height of the SSM’s as much as possible, e.g 80 microns (cracks in SSM’s are not critical)
- keep the solar cell thickness at a reasonable height (e.g. 200 – 250 microns)
- use a special cover glass design (Figure 10-6)
- slides to be slanted at edges
- coating of slanted ares with thermally highly reflective material (e.g. white paint)
- enlarge area across gap and partly across the neighbouring SSM

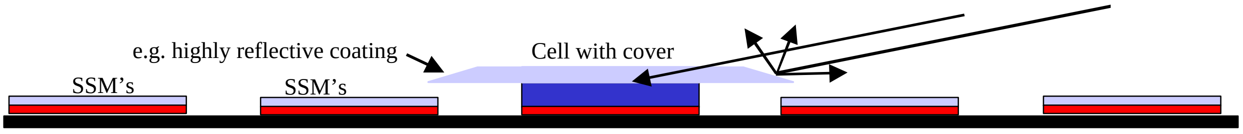


Figure 10-6: Proposed concept to prevent local hot spots under small Sun incidence angles

As soon as a specific design has been defined in detail for the orbit array, it is essential that a detailed thermal analysis is being performed and verified by test on representative samples. Due to the extreme environment and large SSM/cell ratio this specific SA design requires special attention in the detailed design development activities and a detailed verification.

The EOL power generated by this solar array is 500 W in the maximum power point at any time during the observation phase (PMM tracker is used in the PCU). A detailed EOL power profile as function of the Sun distance is given in Figure 10-7. The deviation from the cosine law at small Sun incidence angles has been taken into account.

As for the cruise SA the EOL power predictions for the orbit SA considers 1E15 1MeV el/cm^2 and no further losses.

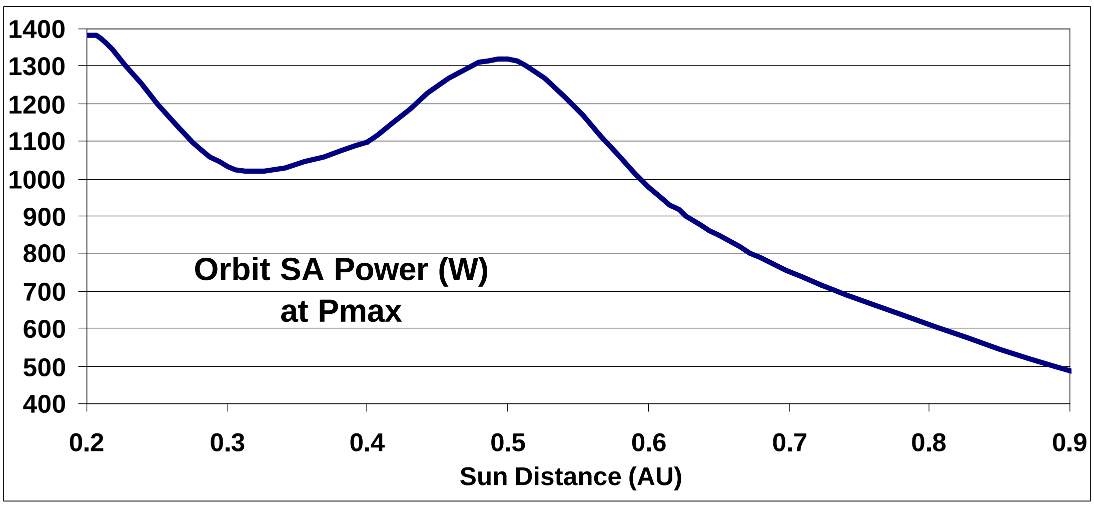


Figure 10-7: Orbit solar array power profile between 0.21 AU and 0.89 AU (EOL radiation fluence)

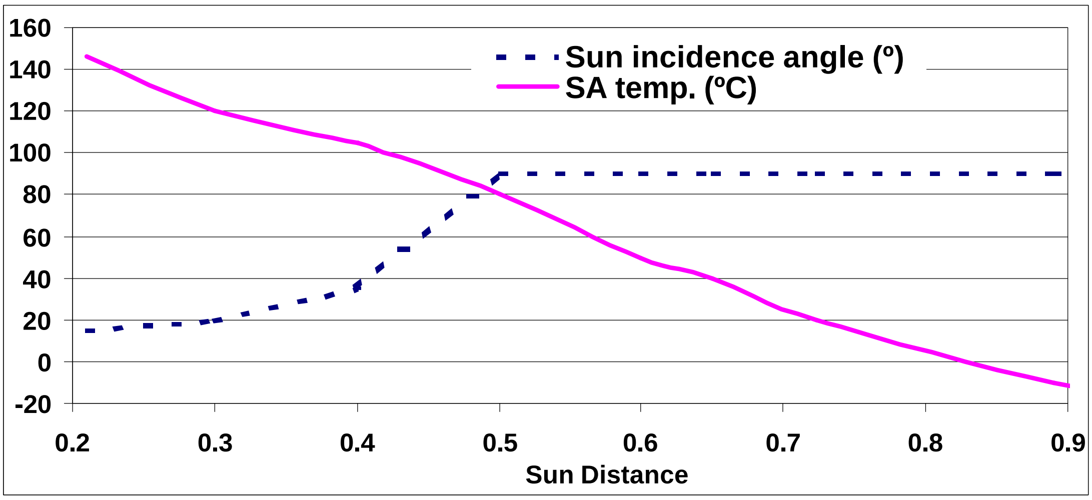


Figure 10-8: Orbit solar array temperature profile and Sun incidence angle on the SA between 0.21 AU and 0.89 AU

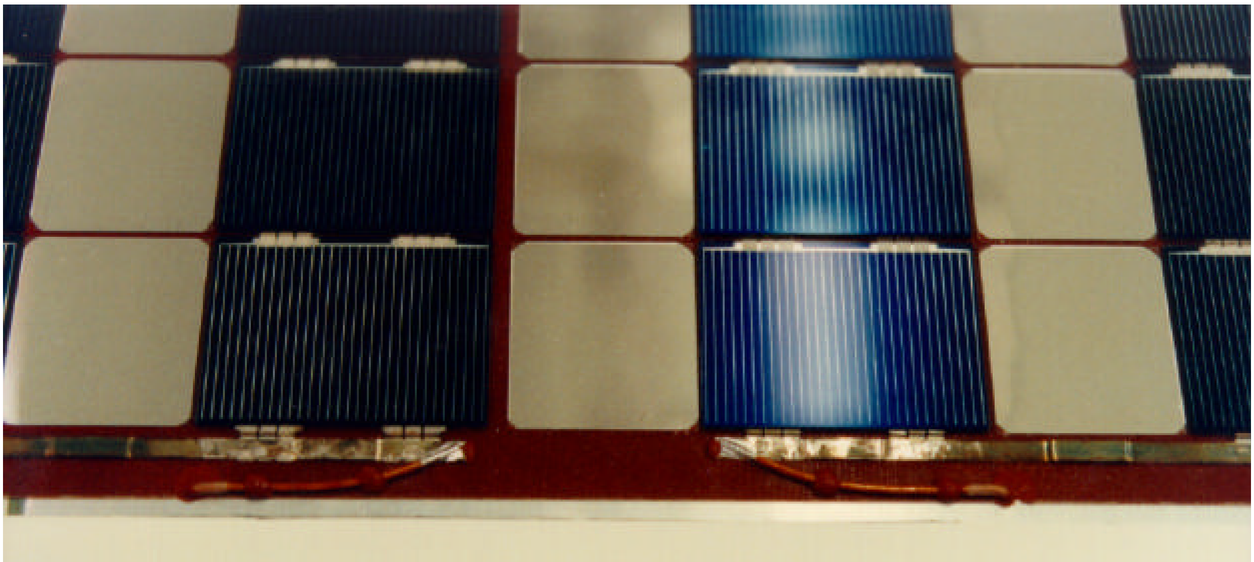


Figure 10-9: Detail of the ISO solar array panel showing the concepts of mixed solar cell / SSM layout.

11 THERMAL CONTROL

11.1 Introduction

The extreme environments encountered by the S/C throughout the whole mission mainly drive the thermal design of Solar Orbiter. On one end, the S/C is orbiting the Sun at distances as close as 0.21 AU. On the other end, the S/C is flying as far as 1.21 AU from the Sun. Another important factor is the electrical propulsion that generates intermittently, i.e. during firing, a considerable amount of heat inside the S/C. Figure 1 gives an overview of Solar Orbiter mission phases. Therefore the thermal design has to accommodate a wide range of heat load levels and locations.

The thermal design of conventional type (selected surface finishes, MLI, non-operational heaters…) uses proven hardware supplemented by the use of thermal louvred radiators and heat pipes on the sunshield to cope with the extreme environmental conditions. The thermal design of critical items such as the cruise and the orbiter solar arrays, the HGA, the payload CCD (low temperature) radiators, the sunshield and the S/C (ambient temperature) radiators has been investigated. As a result, the associated heater power and mass budgets have been established. The trajectory figure in the Mission chapter shows the relative position of the Earth, Sun and Spacecraft throughout the mission.

11.2 Thermal Requirements

At the tart of the project a set of requirements, mainly on temperatures, have been established. The thermal design has been geared to fulfil all those requirements. The following table gives the main design goals:

Item	**(ºC)** **Operating T**	**(ºC)** **Non-Operating T**
Cruise SA Solar Cells	-80 / +120	-100 / +130
Orbiter SA Solar Cells	-80 / +150	-100 / +160
Electronic Units	-20 / +50	-30 / +60
CCD radiative sink	< -140	na

Table 11-1: Temperature Requirements

11.3 Environmental Conditions and Power Dissipations

11.3.1 Solar Environment

The solar intensity at 1 AU, the average Sun-Earth distance, is SC=1371 W/m^2 and varies like 1/d^2, d being the Sun-S/C distance (c.f. Table 11-2). During the cruise phase the Sun-S/C distance varies between 0.25 AU and 1.21 AU. The cruise phase ends by the jettison of the cruise solar arrays. From that time, the S/C enters the observation phase and the Sun-S/C distance is changing from 0.21 and 1.21 AU.

(AU) **d**	**(SC)** **Solar Intensity**	**(W/m^2)** **Solar Intensity**
0.12	22.7	31088
1.00	1	1371
1.21	0.7	936

Table 11-2: Solar Intensity v.s. Sun-S/C Distance

11.3.2 S/C Radiators Sink Temperature

More an induced than a direct environment, the radiative sink temperature is of prime importance to estimate the required radiative area of the S/C radiators. Those radiators are located on the +/- Y side panel of the S/C structure as depicted in Figure 3. In the case of Solar Orbiter, the proximity of the sun combined with the rotation (slew) of the solar array as a function of the solar distance, induce higher sink temperatures than usual like for a S/C earth distance from the sun. To this aim, dedicated models of the S/C for the cruise and the observation configurations have been built. A local model of the S/C with the cruise or the orbiter solar arrays has been developed to derive the sink temperature given in Table 11-3.

Mission Phase	**SA in view of Radiator**	**(AU)** **d**	**(ºC)** **T_{sink}**
Cruise	Cruise	0.33	-86
Cruise	Cruise	1.21	-130
Observation	Orbiter	0.21	-80
Observation	Orbiter	0.89	-100

Table 11-3: S/C Radiator Sink Temperature

11.3.3 Payload CCD Radiator Sink Temperatures

Another induced environment is the sink temperature of the payload instrument CCD radiators. This type of radiator is accommodated directly on the payload instrument body facing the +/-Z directions (c.f. Figure 11-1). As for the S/C radiators, a dedicated model of the S/C has been developed to compute the sink temperature of the CCD radiators. . It was assumed that during communications with the Earth, the CCD would not be operated because of the hot radiative sink temperature induced by the HGA dish. After the ejection of the cruise SA, the observation phase can start. During this phase, only the corners of the orbiter SA are visible from the CCD radiators. It was assumed that the orbiter solar arrays were at 75º to the sun and their temperature was set to 150 ºC in the model. This temperature is the maximum operating temperature (by design) of the orbiter SA at 0.21 AU. In reality, the average temperature of the orbiter SA is lowered by the presence of Optical Solar Reflector (OSR) tiles (c.f. solar array design). During the observation phase, at the closest distance from the sun (0.21 AU), the sink temperature is just below –140 ºC.

11.3.4 Power Dissipations

Some of the S/C power modes are relevant to the thermal design. They give the maximum and minimum power dissipations (c.f. Table 11-4) inside the S/C. The power dissipation varies by a considerable amount (factor 4.8) between the two extreme Sun-S/C distances. As can be seen the maximum power dissipation occurs at 0.33 AU during the SPT firing period of the cruise phase. On the other hand, the power dissipation is considerably reduced when the S/C is in observation phase.

Mission Phase	SPT Mode	(AU) d	W) Pp/l	W) Psvm	(W) Pspt	(W) Ptotal
Cruise	Firing	0.33	0	861	591	1452
Cruise	Firing	1.21	20	686	276	982
Cruise	Non Firing	0.25	0	559	9	568
Cruise	Non Firing	1.21	20	539	9	568
Observation	na	0.21	102	329	9	440
Observation	na	0.89	50	243	9	302

Table 11-4: Power Dissipations

11.4 Thermal Design Description

11.4.1 Accommodation

The final S/C configuration is represented in Figure 11-1 and Figure 11-2. One of the design major challenges was to cope with the extremely high solar fluxes especially during the S/C depointing required when firing the SPT thrusters. Several combinations SPT thrusters location/firing strategy were traded-off. Finally the SPT thrusters were located on the $+Z_{S/C}$ panel and the firing strategy was tailored to the S/C thermal control needs. The whole concept consists in shading as much as possible any part of the S/C from the sun at the closest sun distance and during the SPT firing of the cruise phase. To this aim, a sunshield has been accommodated at the +X side of the S/C. It extends on the +/-Z sides to make sure that the S/C is shadowed even when the +X axis of the S/C is depointing as much as 10º from the sun. The +/-Z depointing of 10º occurs during the SPT firing at the closest distance to the sun (0.33 AU) during the cruise phase. However, other SPT firings occur at higher angles: as high as 28º at the closest Sun-S/C distance of 0.67 AU. For that particular case, it has been verified that the solar intensity is harmless to the S/C protruding parts (CCD radiators, SPT thrusters, HGA mechanisms…) provided they are insulated with typical MLI blankets.

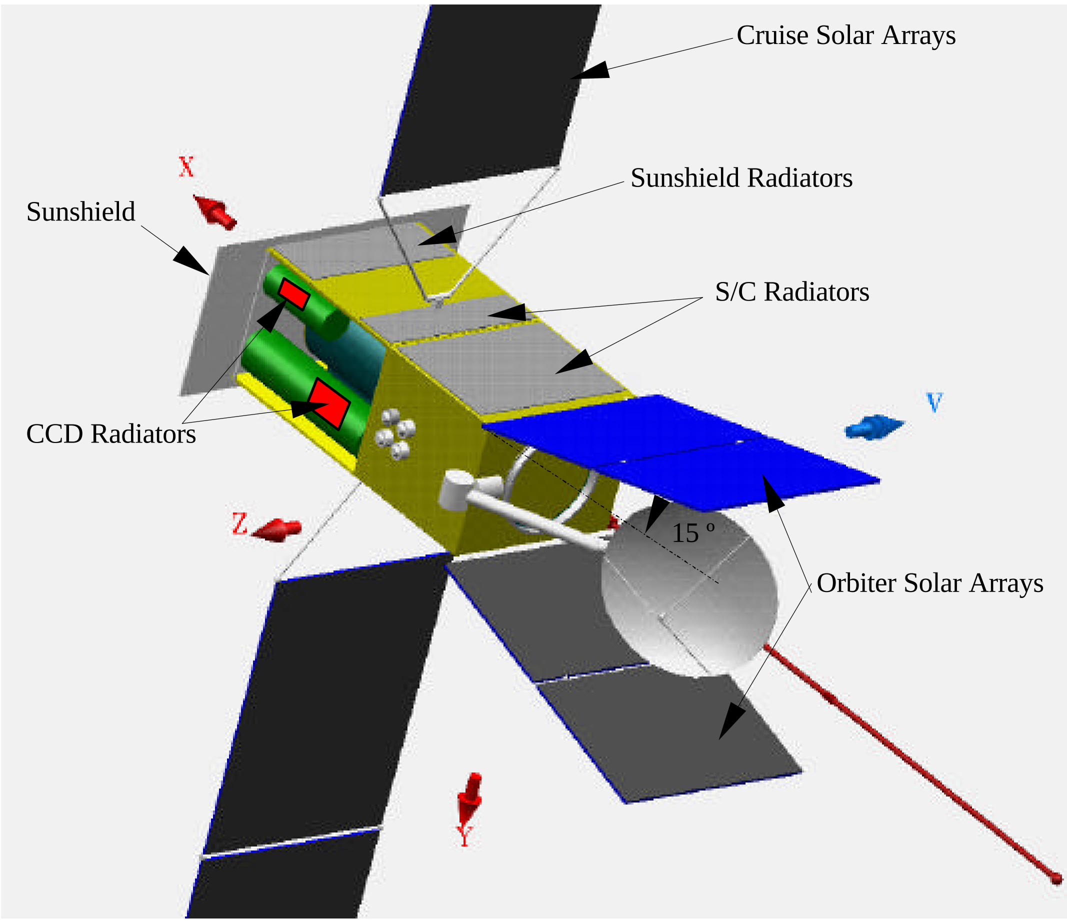


Figure 11-1: S/C Configuration during Cruise Phase (HGA and Orbiter SA are Stowed)

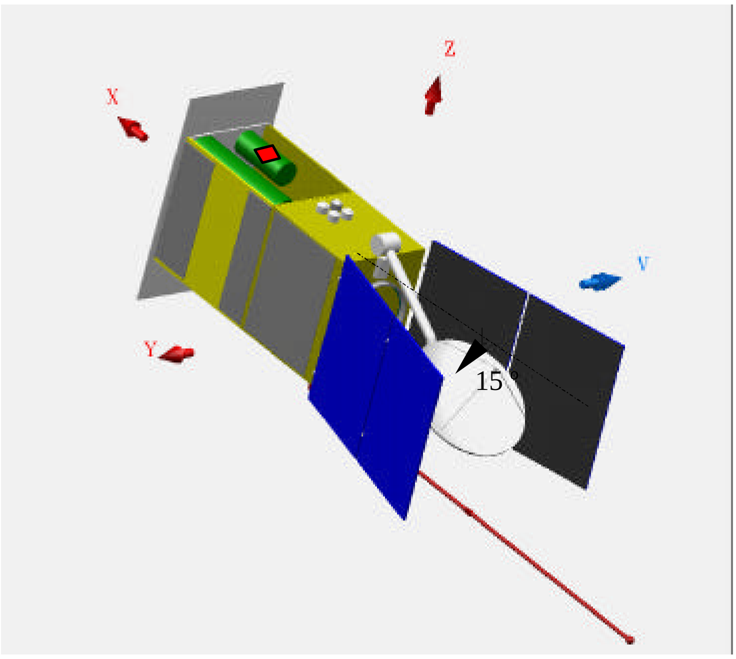


Figure 11-2: S/C Configuration during Observation Phase @0.21 AU

11.4.2 Sunshield and Heat Pipes

The 4.3 m^2 sunshield is supported by a lightweight honeycomb structure. It is made of a stack of 3 titanium foils on top of a standard 15 Kapton/Mylar/Dacron net MLI. The Ti foils are incorporated to survive the high temperatures and to cut the temperature down to an acceptable range (<250 ºC) for the 1^{st} Kapton foil. The outermost Ti foil is white painted that needs to withstand temperatures as high as 490 ºC (Figure 11-3). This is considered as a technological challenge. Behind the structure another 15 layers MLI further insulate the S/C from the remaining flux eventually leaking through the MLI and avoid heat leak during the coldest phases. For the design it was assumed that the overall MLI efficiency must be of the order of 0.01 (ε_{eff}). Any higher value would have a serious impact on the thermal design due to the additional heat leaks.

Property of	α (-)	ε (-)	α/ε (-)
White Paint (BOL)	0.20	0.75	0.27
White Paint (EOL)	0.45	0.75	0.60

Table 11-5: Sunshield Properties

To cool down the sunshield and avoid heat leaks inside the S/C, a 1.7 m^2 dedicated radiators so called “sunshield radiators” are accommodated on +/-$Y_{S/C}$ sides and dump 776 W at 50 ºC at the closet distance to the sun (0.21 AU). The data presented in Figure 11-3 corresponds to a 2.6 m^2 sunshield and have to be scaled for an area of 4.3 m^2. 8 NH_3 variables conductance heat pipes are conducting the heat from the lightweight sunshield structure to the radiators. The Argon filled reservoirs are used to avoid freezing of the NH_3 during the coldest operation phases by pushing the Argon into the condenser. Provision of heat pipes has been made in case the thermal conductance between the sunshield structure and the sunshield radiators is not high enough.

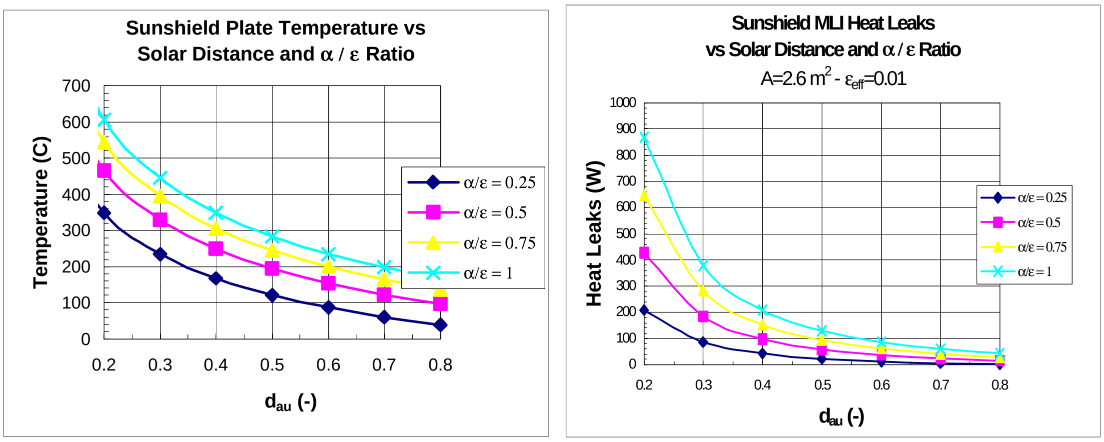


Figure 11-3: Sunshield Temperature and Heat Leaks

11.4.3 CCD Radiators

The payload instruments and their CCD radiators are located on the +/-Z sides, where they are exposed to the coldest radiative sink temperature. The sink temperature of –140 ºC estimated at 0.21 AU, the closest distance to the sun allows lifting nearly 2 W at –80 ºC for a typical 400 cm^2 radiator area.

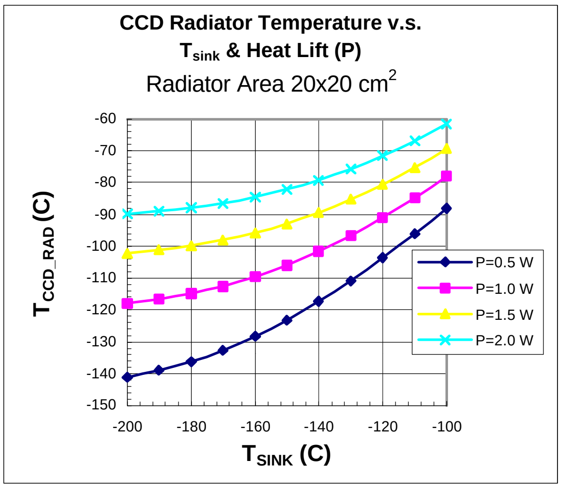


Figure 11-4: CCD Radiators Temperature

11.4.4 S/C Radiators and Louvres

The S/C radiators less demanding in terms of temperature are located on the +/-Y panels and covered with OSR tiles. The electronic units are accommodated on the rear face of the radiators. They will host spacecraft or payload electronic units. Because of the wide range of power dissipations and to minimise the heater power when the S/C is far from the sun (1.21 AU), all the radiators are covered with louvres. The louvres are those developed by STARSYS for the ROSETTA S/C. The properties of those louvres are given below for the open and closed position. The total louvered radiator area required is 4.6 m^2.

Settings	**Open**	**Closed**
ε	0.74	0.14
Temperature	-9 ºC	+5 ºC

Table 11-6: Louvres Properties

11.4.5 Cruise and Orbiter Solar Arrays

Another challenge was the thermal design of the cruise and the orbiter solar arrays. The worst thermo-optical and physical properties of the material constituting both solar panels are given in Table 11-8 and Table 11-8. In addition the reflection law of light has been modified as explained in the Solar Array section.

Property of	α (-)	ε (-)
Unloaded Cells	0.73	0.805
Loaded Cells	0.61	0.805
Kapton @ EOL	0.75	0.75
OSR @ EOL	0.40	0.82
Electrodag 501 @ 150 ºC	n.a.	0.75

Table 11-7: Solar Array Material Properties

11.4.6 HGA and LGA

To lower the temperature of the HGA, it is painted white on both sides. The HGA antenna shall not exceed 140 ºC if it is exposed to the sun at solar distance greater than 0.50 AU. For the closest solar distances, the HGA shall be shadowed by the S/C main body. The LGA thermal design is very challenging because the antenna cannot be shadowed from the sun at all times. The design has to be investigated more in depth.

12 POWER

12.1 Requirements and Design Drivers

The design drivers for power S/S definition have been the following:

- **Optimisation** of **Thermal** Constraints and **High Power** requested by electrical propulsion. This has impact on both Orbit and Cruise Solar Array design, together with the power S/S architecture related to high dissipation path (power conditioning chain to propulsion subsystem).
- **Minimisation of both mass and cost**: i.e. use of cheap and smart solution for batteries (Li-Ion technology instead of NiCd ones), most common schemes and functions which are possibly already available on the market; the last point is especially true for PCU and PDU functional blocks (S3R, MPPT, BDR’s, LCL’s). Solar array technology developed on Mercury.
- **No eclipse during Cruise and Orbit phases**: i.e. small batteries for launch, initialisation and transient peak power demand as well as for contingency purpose. Very low discharge/charge cycles.
- **S/S architecture** derived from **Mercury** with minor modifications (Shunt Regulation)
- **Bus regulation at 50V** to allow use of standard equipment and reduce power losses and harness mass.

12.2 Assumptions and Trade Offs

From the above main design drivers the subsystem architecture that has been selected is a fully regulated bus, with a standard three domains transconductance control loop. This offers the advantage of having a well-known topology together with a wide availability on the market of most of the internal functions. The only difference, with reference to a conventional three domains power system, is the use of different regulator topologies for the Cruise Solar Array and the Orbit Solar Array power. In fact the first uses a S3R regulator while the second a Maximum Power Point Tracking regulator. This solution has been suggested by two main factors:

1. **During Cruise phase the amount of power to be conditioned is very high** due to the presence of the Electrical Propulsion. The maximum amount of power requested by all the main bus users is about 7.5KW. Under these conditions the power dissipation becomes the most critical factor from the thermal control sizing, considering also that the above power figure is requested at 0.33AU (see Power Budget below). For the current baseline, the thermal constraints have been considered as highest priority level design drivers, leading to select the less dissipative topology for the regulator i.e. the S3R.
2. **During the orbit phase only the Orbit solar arrays are available and the requested power is radically lower** than the one during Cruise. With this assumption the reduction of the Solar Array dimension, with consequent reduction of mass and cost, has been considered essential. Use of MPPT techniques allows, in this case, to an optimised overall figure for the power subsystem and spacecraft sizing.

S/C Modes		S/C Requested Power	Remarks
Launch Mode		191.06	MAX
		130.87	MIN
Initialisation Mode		504.15	MAX
		398.35	MIN
Cruise Mode	Firing @ 0.33 AU	7544.12	MAX, power & Thermal S/S sizing
	Firing2 @ 0.96 AU		Cruise SA Sizing Case
	Firing3 @ 1.21 AU		
	Firing4 @ 0.98 AU		
Nominal Observation Mode			
Time Share Observation Mode			
Safe Mode			

Table 12-1: Power Budget vs Mission Modes

12.3 S/S Baseline Design

Referring to the block diagram below, the Power Subsystem is composed of:

- **Cruise Solar Array:** 2 wings of 3 panel each. End Of Life Installed Power ~6.2KW@1AU. GaAs Cell technology
- **Orbit Solar Array:** 2 wings of 2 panel each. End Of Life Installed Power ~500W@0.89AU. GaAs Cell technology.
- **Batteries:** 2 Li-Ion Batteries ~200Wh each.
- **Power Conditioning Unit:** 32 shunt sections to condition the Cruise Array Power, 3 MPPT Modules to condition the Orbit Array Power, 2 BDR's, 2 BCR's, Error Signal generation and Majority Voting Logic, housekeeping telemetry and interfaces.
- **Power Distribution Unit:** 4 Foldback Current Limiter for Stay Alive Lines distribution, 16 Latching Current Limiters for Main User Primary Distribution and 10 High Power Latching Current Limiters devoted to the Electrical Propulsion distribution. The unit will also be equipped with some Heater Driving Switches as necessary.

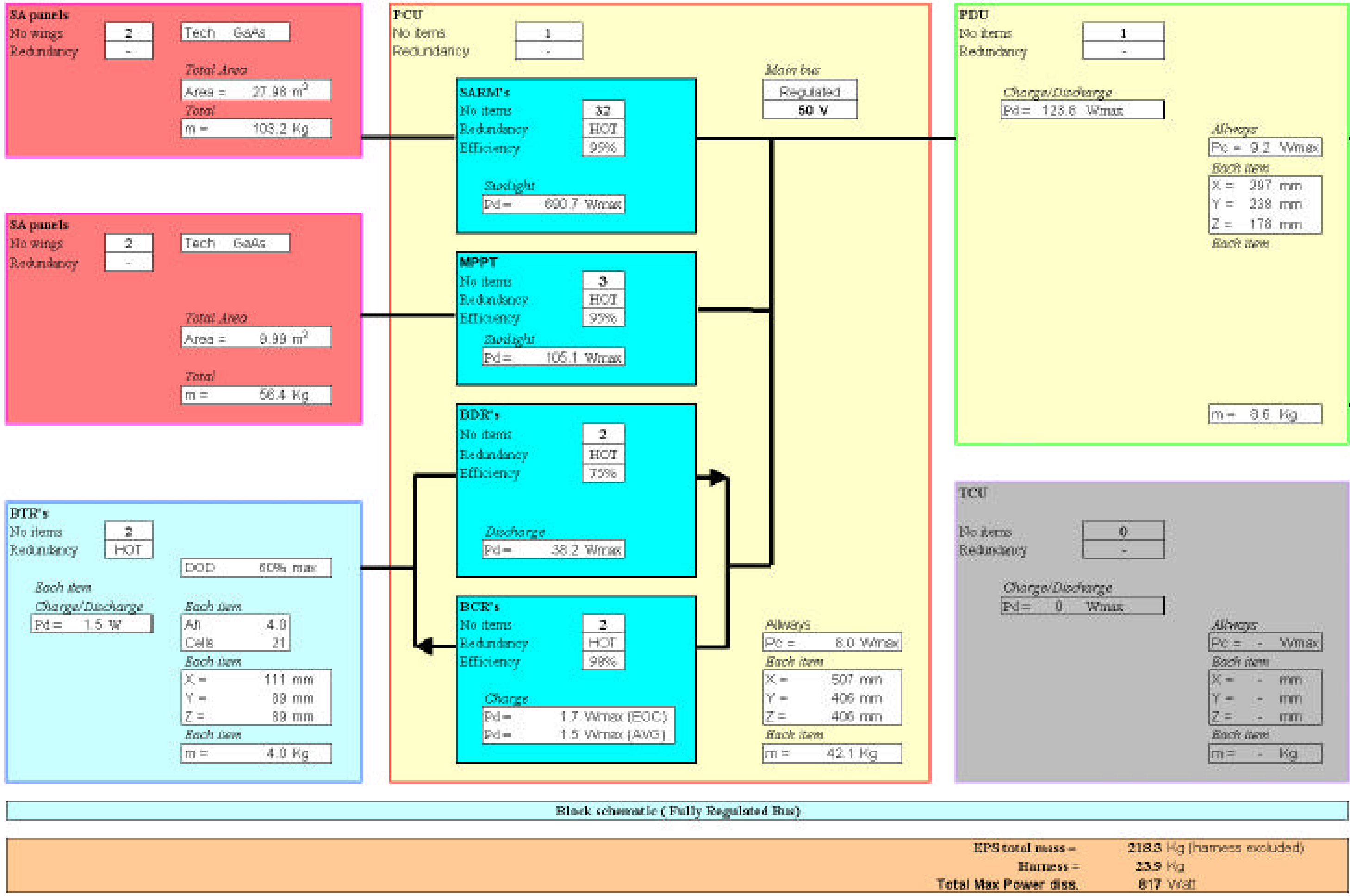


Table 12-2- Power System Block Diagram

A summary of the Power Subsystem performances is given in the following table:

Main bus characteristics			
Bus type	**Fully Regulated**		
MB Voltage =	**50**	V	
Max MB Power =	**7475**	W	
Distribution concept	**Current Limitation Devices**		
Power subsystem characteristic and units			
			Harness excluded
Harness mass	**23.9**	Kg	
Total S/S Mass	**242.1**	Kg	
Cruiser Solar Array			
SA type	**Flat SA**		
Technology	**GaAs**		
Area	**28.0**	m^2	
S.A. Sizing Point	**5576**	W	@ 0.9648 AU
Maximum Load Point	**7475**	W	@ 0.33 AU
Maximum Availability Point	**10830**	W	@ 0.21 AU
Protections			Protection of partial shadowing by diodes
SA mass	**103.2**	Kg	
Orbiter Solar Array			
SA type	**Flat SA**		
Technology	**GaAs**		
Area	**10.0**	m^2	
S.A. Sizing Point	**457**	W	@ 0.89 AU
Maximum Load Point	**457**	W	@ 0.89 AU
Maximum Availability Point	**1616**	W	@ 0.21 AU
Protections			Protection of partial shadowing by diodes
SA mass	**56.4**	Kg	
Power Control Unit (PCU)			
Architecture	**50**	V	Fully Regulated Bus
Functions			MB regulation, battery charge, discharge and autonomous management
Cruiser SA regulator modules (SARM)			Sequential Shunt Regulators (S3R)
Orbiter SA regulator modules (MPPT)			Maximum Power Point Tracker
Number of SARM's	**32**		
Number of MPPT's	**3**		
Battery Charge			From Main Bus
Charging lines switches			2 in series for redundancy
End Of Charge (EOC) strategy			Battery voltage (temperature dependent limit)
BatteryDischarge			By Battery Discharge Regulators (BDR's - one for each battery)
BDR's baseline			SMART
BCR's baseline			BUCK
BDR's efficiency	**90**	%	
BCR's efficiency	**80**	%	
PCU mass	**42.1**	Kg	
Batteries (BTR's)			
BTR's technology	**Li - ion**		
Number of BTR's	**2**		
BTR capacity	**4**	Ah	
Battery cells in series	**15**		
Max DOD	**50**	%	Each battery
BTR mass	**4.0**	Kg	Each battery
Total BTR's mass	**8.0**	Kg	
Power Distribution Unit (PDU)			
Functions			Power distribution to the users (both Platform and Instruments) via current protected switchable (LCL's) or non switchable lines (FCL's), current and status monitoring, heaters switches functions.
Philosophy			Centralised power protection by Latching Current Limiters (LCL's), for all users but CDMU and Receivers, which are protected by Foldback Current Limiters (FCL's).
LCL lines	**16**		2 spare lines taken into account
High Power LCL lines	**10**		15 A each line devoted to the Propulsion Power Lines
LCL characteristics			Distribution in 6 power classes (A to F), trip OFF time 11 to 16 ms, MB undervoltage protection on each LCL line, ON/OFF capability with presettable auto ON or OFF preset. Status and current monitor of each LCL line.
Heaters switches	**8**		Non-redundant
FCL lines	**4**		2 for CDMU; 2 for receivers.
PDU mass	**8.6**	Kg	

Table 12-3: Power System Characteristics

12.4 Critical Areas

One area for further investigation has been identified concerns the possibility of reducing the area (mass and cost) of solar array by about 20%, adopting a Maximum Power Point Tracker conditioning and regulation solution. This option, briefly studied during this phase, should be more deeply investigated. The current baseline uses, for the cruise arrays power in sunlight mode, a conventional S3R regulation. This leads to a better conversion efficiency and consequent low power dissipation constraints. The use of MPPT techniques should provide with an optimised utilisation of the SA installed power. However, due to a lower efficiency of the conditioning chain, this solution increases complexity (and cost) of both the Thermal Control System (already at the state of the art limit with the current design baseline) and the Power Conditioning Unit. A detailed trade-off with the aim of selecting the best compromise between design complexity, mass and cost, should be performed.

13 MECHANISMS

13.1 Design Approach / Heritage

In the following paragraphs the design approach and the main assumptions / heritage for the selected configurations are detailed.

13.1.1 High Gain Antenna Pointing Mechanism

A Two-axis steerable system is needed to point the high gain antenna toward earth for communication purpose (X/Ka band).

The main design driver has been the compatibility with the severe environment dictated by the mission profile, in particular the very high heat fluxes, which can be experienced in sun proximity.

The design strategy has been to accommodate both axes of the pointing mechanism close to the Orbiter body structure, to achieve both good thermal exchange with surrounding structure, and simplify the thermal protection/shielding from direct Sun exposure.

By means of this strategy, conventional operativing temperature ranges are expected for the mechanism, without the need for high temperature material/components technology development.

The above is valid also for the RF rotary joint, which will feed the RF signal through the two axis rotary stages.

In particular the mechanism azimuth stage is largely inside the spacecraft body, whilst the elevation stage is protruding (for mechanical clearance reasons).

The main thermal shielding is provided by the main thermal shield placed on the top of Orbiter structure (the sun pointing face). This prevents direct sun exposure when orbiting close to the Sun. Conventional thermal control is foreseen locally (shielding and MLI blankets), to protect when moderate heat flux (direct exposure when far from Sun) is expected.

Antenna dish spacing, to allow adequate free field of view, is provided by means of a mast structure placed between the pointing mechanism and the antenna reflector itself. The antenna structure is thermally decoupled from the pointing mechanism.

The disturbances, which are introduced when accelerating the reflector structure, are kept under control by means of spacecraft attitude control system.

The Rosetta High Gain Antenna pointing mechanism has been used as reference for the purpose of budgets estimation and technology availability assessment (in Solar Orbiter time frame).

13.1.2 High Gain Antenna Hold Down and Release Mechanism

A three point hold down and release mechanism is foreseen to keep the high gain antenna structure stowed during launch (for obvious envelope and vibration environment reasons).

A pyro actuated system is envisaged, with a conventional approach for launch load transfer through separable surfaces.

No particular problem is expect from potential high temperatures, because the antenna release function will be operated when far from the Sun.

13.1.3 Cruise Solar Arrays

Conventional technologies / products will be used for the Cruiser solar arrays. A spring based deployment system, with damper controlled deployment speed is foreseen.

The whole solar array system (including panels and drive mechanism) will have to be jettisoned for overall mission optimisation reasons. A jettisoning mechanism will be implemented, interfacing both the drive mechanism and the spacecraft mounting panel.

The separation / jettisoning function will be realised by means of three pyro released connection elements, plus spring preloaded ejection devices. Pull apart connectors will allow the separation function for all electrical connections.

13.1.4 Orbit Solar Arrays

Dedicated solar arrays (two panels for each wing) will be implemented for mission orbit phases close to the sun.

Conventional design solutions can be envisaged for the hold down / deployment function (operating in a standard temperature environment).

In addition to the initial release / deployment function, wing rotating capabilities have to be implemented to cope with high thermal fluxes. Based on the maximum allowed temperature on the array panels, each wing will be tilted wrt Sun LOS, thus reducing to acceptable levels the impinging heat flux on the panels.

Stepper motor based actuators can provide the required accuracy for the tilting angle. Thermal protection / shielding from direct sun exposure and good thermal coupling with spacecraft mounting structure will ensure standard temperature ranges also for this actuators.

13.2 Budgets

The estimated mass and power budgets are reported below. Mass figures include (where applicable) the driving electronic mass.

	MASS [kg]	POWER [W Peak]
HGA Pointing / Hold Down-Release	26.0	16.0
Cruise SA Drive	15.0	16.0
Cruise SA Release/Depl. system	16.0	-
Cruise SA Jettisoning	6.0	-
Orbiter SA (release/depl. and actuators)	20.3	12.0
Thrusters pointing (*)	20.0	40.0

Table 13-1: Mechanisms Mass and Power Budget

(*) Not included in the current baseline (Artemis derived figures)

14 PYROTECHNICS

14.1 Requirements and Design Drivers

The need to limit system mass and volume leads to the proposal to use pyrotechnic devices for all single-cycle functions on the Solar Orbiter mission. Twenty-seven such single-cycle functions are needed in the propulsion system and to release appendages. The propulsion system tanks will be sealed during launch and will be opened to supply the thrusters. The solar arrays will be stowed for launch and then deployed as soon as possible to provide power. The High-Gain Antenna and a number of Low-Gain Antennas will be stowed for launch; these will be released as soon as possible after launch to establish communications. The Cruise array is to be jettisoned when the electric propulsion is no longer needed.

The requirement to use off-the-shelf equipment is satisfied by the qualified components, which are available.

14.2 Assumptions and trade-offs

Pyrotechnic release-nuts and valves exist which are suitable for the loads and dimensions envisaged. For reliability, each actuator is equipped with redundant initiators each controlled and supplied by independent circuits.

The short duration of current pulse needed to fire a pyrotechnic device means that the power demand is negligible in comparison to other power users on board. Overall subsystem mass is lower in comparison to other technologies due to smaller power supplies and thermal control equipment. Although the release using pyrotechnic devices produces shock, the subsystem equipment is robust enough to sustain this disturbance without degradation.

Apart from Cruise Array jettison, all the functions are executed soon after launch. Provided this jettison is done before the spacecraft is too close to the Sun, no specific thermal protection will be needed.

14.3 S/S Baseline Design

The susbsystem comprises all the pyrotechnic actuators, their dedicated supply wiring, switching to avoid premature firing and pulse-shaping electronics taking power from the main bus. From the safety switches the wiring to the initiators will be screened twisted pairs to avoid susceptibility to radio-frequency and electromagnetic pick-up. Unique connectors are used to avoid connection to the wrong circuits. Limitations on survival temperature of the pyrotechnic material at 90 Celsius, require that this last function is performed before the spacecraft approaches too close to the sun. Jettison at Venus will ensure that such temperatures are avoided.

15 ATTITUDE AND ORBIT CONTROL

The basic functions of the Attitude and Orbit Control Subsystem (AOCS) are:

- To accurately point the spacecraft optical reference axis Xo to the Sun, except during SPT-firing mode, with a maximum pointing stability of 1 arcsec (3σ, half-cone angle) over 15 minutes during the nominal observation mode
- To keep the thrust axis of the SEP engines aligned to the ΔV direction during SPT-firing mode with a maximum deviation of less than 1°
- To maintain the spacecraft into a safe sun-pointing attitude using a minimum of on-board resources while ensuring power generation and ground communication

The spacecraft attitude is three-axis controlled from the release on orbit and throughout the mission. After initial rate reduction and sun acquisition based on gyroscopes, coarse sun sensors and thrusters for actuation, an initial attitude (nominally sun pointed) is acquired using the autonomous star pattern recognition of the star tracker. The routine operations of the Solar Orbiter mission will all be performed in the Normal Mode. This mode includes all the functionality necessary to perform the communications or the nominal science observations (Sun pointing), and the slew manoeuvres necessary between all these operations. The attitude measurement is performed with the autonomous star tracker, while a set of four 4 Nms reaction wheels in a pyramid configuration is used as primary actuator. These wheels have been selected due to system mass design constraint, and frequent wheel off-loading is required during nominal science observations. The ΔV and insertion manoeuvres are covered by the SPT-firing mode.

A single-axis linear covariance analysis has been performed to assess the minimum AOCS error contribution, mainly star tracker noise and wheel quantization, to the instrument pointing stability budget. For a star tracker noise of 4 arcsec (1σ), resp. 1 arcsec (1σ), the contribution of the AOCS errors, Table 15-1, amounts to 1.26 (resp. 0.42) arcsec which is above the pointing stability requirement. Therefore, either a better noise performance star tracker or a relaxation of the system pointing stability requirement is needed.

	Cruise Mode	Observation Mode	Comments
Attitude Control			
• STR noise + RW quantization	0.83 / 0.30	1.17 / 0.42	Covariance analysis (STR:4 arcsec/1arcsec)
• delay	0.10	0.10	Allocation
• env. disturbances	0.10	0.10	Allocation
Dynamic Interaction			
• high frequency disturbances (RW)	0.10	0.10	Allocation
• thermo-elastic	0.40	0.40	Allocation
• others	0.20	0.20	Allocation
Pointing Stability (15 min)	0.96 / 0.56	1.26 / 0.67	RSS - single axis, 1σ
Requirement (single axis)	0.29	0.29	Assumptions: 1 arcsec over 15 min specified half-cone angle at 3σ

Table 15-1: Pointing Stability Error Budget with STR noise of 4 arcsec / 1 arcsec (1σ) in the S/C body reference frame

The AOCS hardware architecture, Figure 15-1, is inherited from on going ESA scientific missions. The Control and Data Management System (CDMS) includes a dedicated processor for the AOCS s/w and the high level monitoring functions required during the ΔV and insertion manoeuvres for the autonomy and fail-operational capabilities. Considering the programmatic design constraints, the AOCS uses a lot of units already developed and validated for other programmes. In the baseline, the Inertial Measurement Unit (IMU), the Sun Acquisition Sensors (SAS), the Star Tracker (STR), and the Reaction Wheels Assembly (RWA) are Mars Express and Smart-1 recurring units, leading also to a good level of reuse in the AOCS interface unit (AIU). The total mass and maximum power of the AOCS baseline, Table 15-2, are 25.8 kg (without margin, which is applied at system level) and 68.4 W respectively.

A monopropellant (hydrazine) system has been selected as reaction control system (RCS) for spacecraft attitude control after separation and safe mode, and reaction wheel off-loading. It consists of 2x6 5 N thrusters, one hydrazine tank, valves, piping, heaters, and insulation. The thruster configuration has been finalised on the basis of minimised cosine losses, minimised plume impingement, and best moment arm around the spacecraft Y-axis. The total momentum required to fulfil the Solar Orbiter mission, inc. extended mission and 50% AOCS design margin (*but excluding the potential impact of S/C c.o.m. shift during the cruise phase),* has been estimated to be 16000 Nms which corresponds to 15.5 kg of hydrazine.

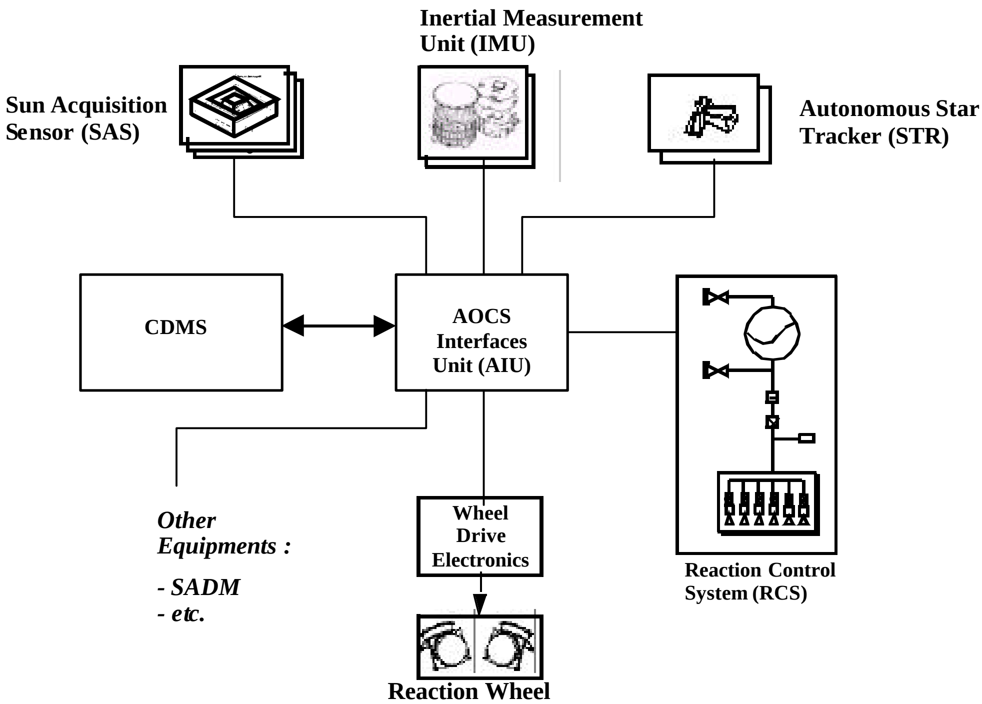


Figure 15-1: AOCS hardware architecture

AOCS Unit (baseline)	Nb	Mass/unit (Kg)	Power (W)	Main Features	Supplier
Star tracker (STR), inc. baffle	2	0.63	0.6	Accuracy: 4” (3σ)	DTU
Star Tracker Electronics (STRE)	2	1.17	7.0	-	DTU
Gyro (IMU)	2	4.10	34.0	Bias: 0.05°/hr (1σ)	Honeywell
Sun Acquisition Sensor (SAS)	3	0.23	-	Accuracy: 2° (2σ)	TPD-TNO
Reaction Wheel (RW)	4	2.55	6.7	4 Nms / 0.02Nm	Ithaco
Wheel Drive Electronics	1	3.12		-	Ithaco

Table 15-2: AOCS Baseline Hardware units

16 DATA HANDLING

16.1 Mercury Mission Heritage

The data handling of the Solar Orbiter spacecraft has been derived from the one proposed for the Mercury Cornerstone mission. In this respect some assumptions had to be made because the data available to the study team were limited. It is therefore possible that the mass and power values will change when the cornerstone mission will mature, especially considering that those values are largely based on the today's spacecraft technology, that will follow a natural course of improvement in the following years.

The Data Handling architecture of Mercury orbiter is composed by CDMS, AIU , payload RTU and spacecraft RTU. The CDMS is the core of the data handling, it encompass the computer, the mass memory and the telemetry and telecommand electronics. The AIU (AOCS Interface Unit) and is similar to an RTU but directly connected to the CDMS; it interfaces the computer with the different sensors and actuators of the AOCS. The payload RTU and spacecraft RTU are instead connected with the computer via the spacecraft bus and provide interfaces with all the other specific interfaces either to the payload or to the spacecraft.

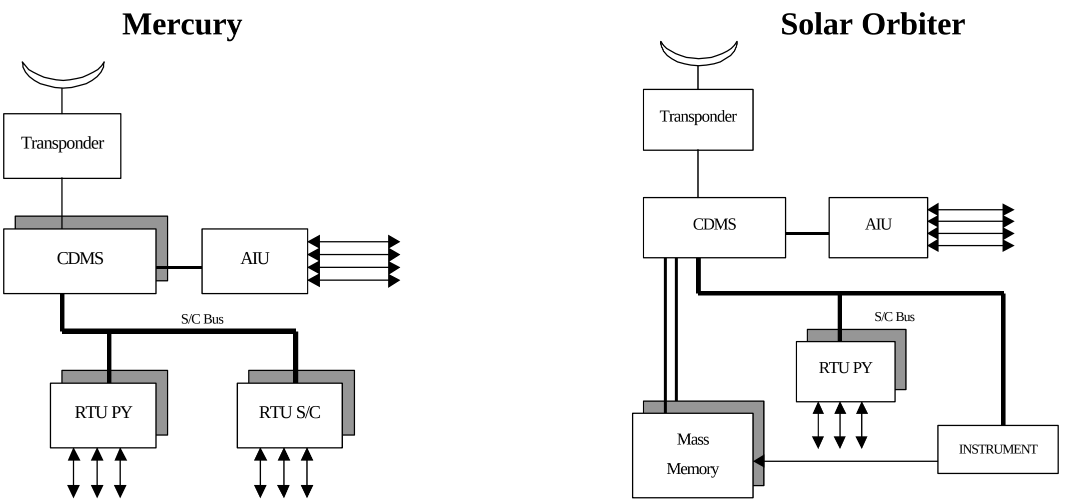


Table 16-1 : Mercury and Solar Orbiter DHS

The figure above shows the diagram of the data handling as it appears in the Mercury phase A study report. The Solar Orbiter data handling differs on the following elements: the CDMS and the S/C RTU. The CDMS of Mercury contains the Mass Memory but in Solar Orbiter this unit is too large to be hosted in the same box. For this reason an external redundant unit has been created to implement the Mass Memory function. The rest of the CDMU remains the same, assuming that the separation from the Mass Memory will require only software changes.

The final architecture of the Solar Orbiter data handling is shown on the right hand side of the above figure.

16.2 Solar Orbiter CDMU

The Solar Orbiter CDMU is derived from the one of Mercury cornerstone, with the major modification of Mass Memory separation from the unit box. The microprocessor memory is identical since we have no evidence at the moment that the Solar Orbiter SW will have larger size than the Mercury one. Other modifications such as the different clock for the telemetry data rate have not impacted on the overall CDMU architecture and will not be listed in details, but it is important to stress that the impact will also depend from the flexibility of the final Mercury CDMU design. The main elements of the CDMU are:

- ERC32 microprocessor and memory
- Transfer Frame Generator
- Telecommand decoder

The figures of mass and power given for the Mercury mission includes Mass Memory and the rest; therefore the following estimation was made in order to derive the Solar Orbiter figures:

	Mass	**Power**
Original Mercury CDMU	14.00	25.00
Estimated Mass Memory	4.00	10.00
Solar Orbiter Reduced CDMU	10.00	15.00

Table 16-2: DHS Mass and Power Comparison

16.3 Solar Orbiter Mass Memory Unit

The major challenge for the Solar Orbiter Data Handling resides in the mass memory: the mission profile foreseen long periods where the S/C is collecting science data and periods where it transmits them to Earth. The size of the mass memory is so large, compared to the Mercury mission, that it requires the development of a new unit. In estimating the characteristic of the unit consideration was given to the fact that the memory device technology is quickly improving and by the time of Solar Orbiter the present prototype memory silicon will be available. The newly developed 1 Gbit memory has been taken into account as the baseline for the mission starting in the next 4 years, since these devices are already available in the commercial market as prototypes. Considering advanced technology device the Mass Memory unit can be realised within reasonable mass and power; 256 Mbit memory chip technology is used the power consumption increases by almost a factor three.

In the estimation of the power consumption has been assumed that the nominal Mass Memory unit is on while the redundant is off. This arrangement might create problems to the mission in case of Mass Memory failure, since this will imply losing a large portion of the overall mission science if this happens just before the earth downlink phase. The solid state Mass Memory is able to suffer internal failures without the complete lost of functionality or data, and only a failure to a few vital parts can cause the complete loss of the unit. For this reason it is estimated that the probability of complete failure is acceptable. An alternative safer strategy is to keep both Mass Memory units on and accumulate science data on both at the expense of doubling the power consumption.

Average Orbit Time		150 *days*			
Perihelion (High Science Rate)		26 *days*	*Aphelion (Low Science Rate)*		124 *days*
Instrument	**Status**	**Data Rate**	**Instrument**	**Status**	**Data Rate**
Spectrometer Package	On	63000 *bps*	Spectrometer Package	Off	0 *bps*
Particles Package	On	11500 *bps*	Particles Package	On	11500 *bps*
Housekeeping	On	1000 *bps*	Housekeeping	On	1000 *bps*
		75500 *bps*			12500 *bps*

Data accumulated in perihelion	169.60 *Gbit*
Data accumulated in aphelion	133.92 *Gbit*
Max theoretical data volume	309.59 *Gbit*

Selected Mass Memory Size	**246 *Gbit***
Number Boards	5
Redundant	Yes
Width	260 *mm*
Height	188 *mm*
Length	262 *mm*
Mass (not including margins)	**6.95 *0.00***
Total Power Consumption	**20.48 *Watt***

Table 16-3: Mass Memory Characteristic

16.4 Solar Orbiter Remote Terminal Unit

In the mercury mission the Data Handling part was equipped with two RTUs to connect the payloads and the spacecraft. In the case of Solar Orbiter, due to the simpler architecture of the spacecraft, only one RTU has been selected, with an increase in its capabilities. As a reference the Mercury RTU has been taken and mass and power increased in proportion to the estimated increase in connections. We shall assume that the payloads shall be able to connect themselves to the S/C or reserved bus and to act as intelligent units without deep involvement of the on board software and S/C harness.

Original Mercury RTU	**Power**	**Mass**
	7.00 Watt	8.00 Kg
Estimated increases	50 %	35 %
Solar Orbiter RTU	**10.50 Watt**	**10.80 Kg**

Table 16-4: Solar Orbiter RTU Characteristics

16.5 Solar Orbiter AOCS Sensors and Actuators Interface

The Mercury and Solar Orbiter spacecraft are similar as far as the AOCS sensor and actuators are concerned. In this respect only minor modifications of the present AOCS interface is expected. Therefore, though the unit will require some engineering work, the number and type of interfaces remains the same and therefore mass and power are taken from the Mercury baseline.

Mass	**7.00 Kg**
Power	**5.00 Watt**

Table 16-5: AIU Characteristics

16.5.1 Software

As for the rest of the Data Handling system the maximum reuse of the Mercury software shall be pursued; in this respect the two missions have much in common as far as manoeuvres and operational modes are concerned. The software can be broken down into a number of standard packages valid for both the spacecraft, and a number of custom design software packages that are modifications of existing SW or new.

Standard applications include both the operating system and typical on-board activities as TC Handler, TM Handler, Time tag and others. Since no software sizing figures were available from Mercury, estimates had to be made that must be considered at this stage to be very provisional. The largest modification on the software is likely to come from the AOCS where it was assumed that the software has to be developed. The AOCS has the following modes:

Safe Mode	Entered when the spacecraft goes into safe mode and is the same regardless we are close or far from the SUN
Navigation Mode	Used when the spacecraft is travelling toward the sun and mostly used for SEP thrusting.
Fly-by Mode	entered in proximity of a fly-by operation; it involves both the data-handling and the AOCS.
Observational Mode	used during the high pointing accuracy data acquisition
Telecommunication Mode	used to point the antenna to download data to earth.

The modification w.r.t. to the original Mercury software package is estimated around 40% of the overall software.

16.6 Sub-System Budgets

	Mass (incl. redundant)	Power worst case
CDMU	10.00 Kg	12.00 Watt
MM	13.90 Kg	20.48 Watt
RTU	10.80 Kg	10.50 Watt
AOCS	7.00 Kg	5.00 Watt
Harness	10.00 Kg	
Total	**41.70 Kg**	**47.98 Watt**

Table 16-6: Sub-System Budgets

17 TELECOMMUNICATIONS

17.1 Telecom Requirements and Design Drivers

17.1.1 Communications Scenarios

Two types of scenarios are identified from the communications point of view:

- Cruise Phase.

This phase takes 1.86 years and a distance up to 2AU has to be supported. The downlink data consists on housekeeping information (up to 1kbps). Communications via wide beam-width Low Gain Antennas (LGA) at X-Band is consider as baseline for both, up and downlink. If required, a High Gain Antenna (HGA) could be pointed toward the Earth and in this case a careful co-ordination with the propulsion activities and attitude manoeuvres would be required.

- Nominal + Extended Phases.

These phases are similar from the communication point of view and last a total of 5.16 years. A total of 12 orbits around the Sun are covered with average figures of 0.2AU Perigee and 0.9AU Apogee. The distance spacecraft-Earth will change with every orbit (from 0.3 up to 1.8AU) , and consequently the downlink data rate that can be supported will vary. Each orbit takes 150 days (average). Sciencific observations at 74.5kbps are performed during 30 days (max) when the spacecraft is close to the perihelion and up to 196 Gbits (including HK) will be stored on board for further download. The thermal environment existing at a distance to the Sun below 0.5AU may damage the HGA, and only for ranges above this, is the high rate link via HGA feasible (approx 110 days/orbit). Out of the 30 days of observations at 74.5kbps, scientific observations at a lower rate (11.5kbps) will take place. When possible this data will be downlinked in real time, otherwise stored on board and dumped later.

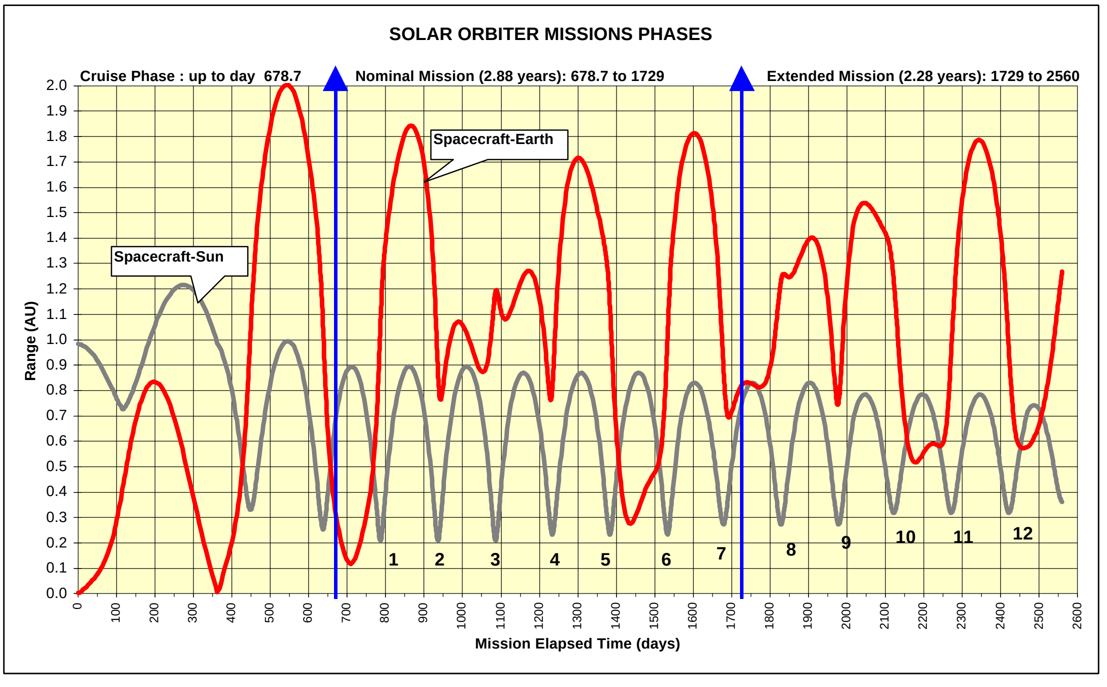


Table 17-1 - Solar Orbiter Range Profile

17.1.2 Ground Segment Requirements

The telecommunication links will be established in X and Ka Band. At Ka Band the antenna gain is up to 12 dB higher than at X Band and even though the free space losses are also increased, higher data rates can be supported.

The baseline station considered for the study is a 35meter ground station with X/X-Ka capability (as the one foreseen in Perth –Australia). The requirements are as follows,

- Uplink : 107dBW EIRP (20kW X-Band Klystron)
- Downlink: G/T = 56.0dB/K @ 10deg/Ka-Band (including pointing losses),
- G/T = 50.1dB/K @ 30deg/X-Band
- Station manned for 8 hours/daily

17.1.3 Transponder requirements

- Telecommand: X-Band receiver, modulation PCM/BPSK subcarrier /PM
- Telemetry: X and Ka transmitters with 20Watts power amplifiers. Modulation with Turbo Coding ½. For symbol rates below 60ksps (operations via LGA), modulation PCM/BPSK subcarrier /PM. For symbol rates above 60kbps (operations via HGA), modulation SPL-PCM/PM or BPSK.
- Ranging only will be performed when supported by the link budget.
- RF distribution unit will interface the transponder(s) with the antenna subsystem. The length of the Solar Mission calls for a fully redundant telecom system, except for the antennas.

17.2 Subsystem Baseline design

The baseline design of the communication subsystem is presented in the figure below:

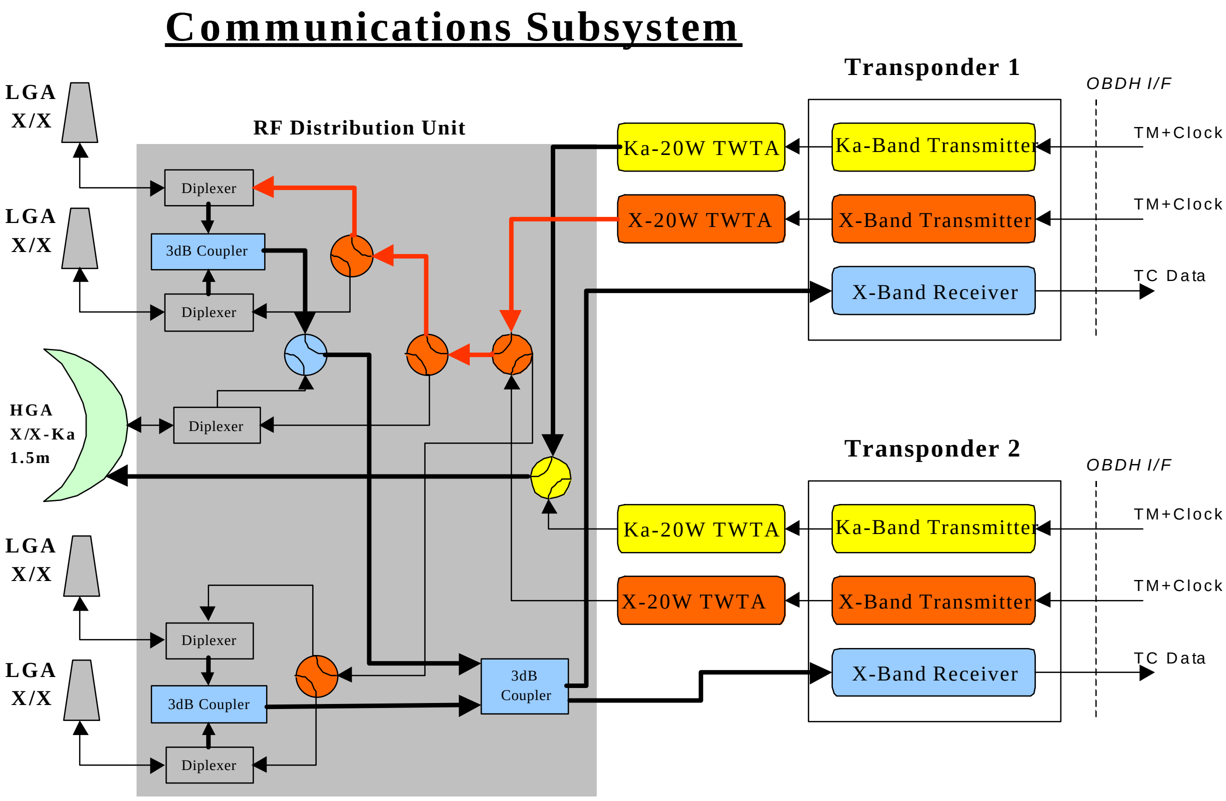


Table 17-2 –Communications Sub-system Block Diagram

The principal link for data return is a 1.3m High Gain antenna operating at Ka Band radiating 20Watts RF power to the 35m ground station. This antenna also supports X-Band downlink and this mode may be useful when Ka-Band link performance is highly degraded due to atmospheric conditions.

The TC uplink will be always in X-Band. When operating via the LGAs, the input signals of all the four antennas are combined at the receiver(s) input. The uplink via HGA is selectable with a switch.

17.3 Telemetry and Telecommand Rates

(Clear sky conditions considered in budgets)	TELECOMMAND	TELEMETRY
LGA X / X	100 bps @ 0.4 AU; 4 bps @ 2AU	60 bps @ 0.4 AU; 4 bps @ 0.9 AU
HGA X / X	4 kbps	170 kbps @ 0.6 AU; 16 kbps @ 2 AU
HGA X / Ka		750 kbps @ 0.6 AU; 98 kbps @ 1.9AU

Table 17-3: Telemetry and Telecommand Rates

Note that for ranges Earth-satellite greater than 0.9AU, telemetry downlink should use the HGA.

17.4 Downlink Approach

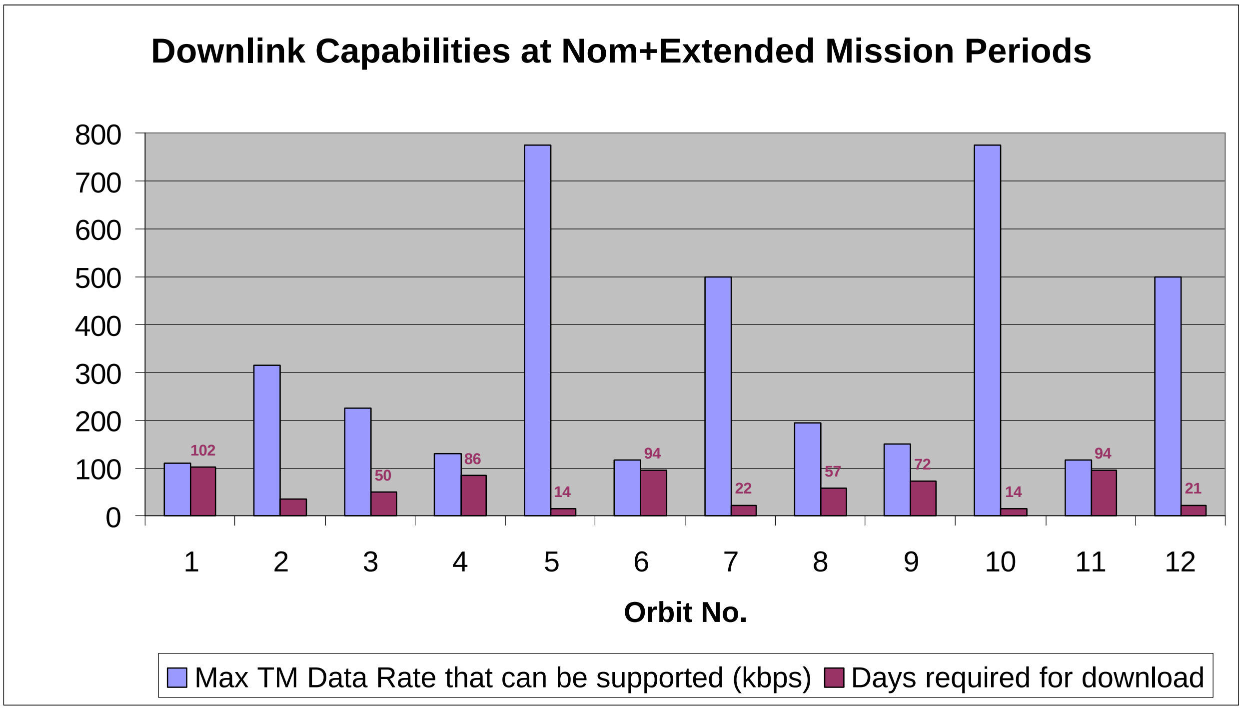
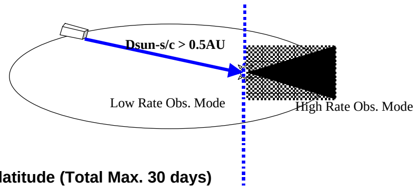


Table 17-4 – Graph of Downlink capabilities during Nominal and Extended mission.

SOLAR ORBITER DATA DOWNLOAD BUDGET: NOMINAL PHASE

Ground Station: PERTH 35m, Ka-Band D/L, Clear Sky (95% of the time)
Spacecraft: SOLO with HGA, 20W Ka-Band Tx, 0.1deg pointing losses

Observation Mode	Full Rate	Low Rate
Scientific Data (bps)	74500	11500
Housekeeping Data	1000	1000
On-Board Data Rate Generation (bps)	**75500**	**12500**



DOWNLINK REQUIREMENTS:

(a) High Rate Observation Mode occurs for +/- 5days at perihelion, higher and lower S/C latitude (Total Max. 30 days)

(b) Low Rate observation mode occurs when High Observation Mode not active

(c) TM Downlink via HGA occurs for a distance Sun-S/C greater than 0.5AU

	NOMINAL PHASE						
ORBIT NUMBER	**1**	**2**	**3**	**4**	**5**	**6**	**7**
RANGE SUN-S/C < 0.5 AU (No. of Days)	38	38	38	38	38	38	38
RANGE SUN-S/C > 0.5 AU (No. of Days)							
High Gain Antenna Downlink (No. of Days @ 95% avail)	**105**	**105**	**105**	**104**	**104**	**103**	**103**
Max Range (AU)	1.84	1.10	1.30	1.70	0.50	1.80	0.85
Max TM Data Rate that can be supported (kbps)	110	315	225	130	775	118	500
FULL RATE OBSERVATION @ 75.5 Kbps (No. of days)	30	30	30	30	30	30	30
Data Stored On-Board during Full Rate Obs (Gbits)	196	196	196	196	196	196	196
LOW RATE OBSERVATION @ 12.5 Kbps (No. of days)	119	119	118	117	117	116	116
Data Stored On-Board during Low Rate Obs (Gbits)	129	129	127	126	126	125	125
DATA VOLUME GENERATED IN ONE ORBIT (Gbit)	324	324	323	322	322	321	321
NO. OF DAYS REQUIRED TO DOWNLOAD THE OVERALL DATA VOLUME AT MAX DATA RATE (8h/day G/S support)	**102**	**36**	**50**	**86**	**14**	**94**	**22**
Margin (No. of Days)	3	70	55	18	89	8	80

	EXTENDED PHASE				
ORBIT NUMBER	**8**	**9**	**10**	**11**	**12**
RANGE SUN-S/C < 0.5 AU (No. of Days)	38	38	38	38	38
RANGE SUN-S/C > 0.5 AU (No. of Days)	108	100	105	106	96
High Gain Antenna Downlink (No. of Days @ 95% avail)	**103**	**95**	**102**	**102**	**102**
Max Range (AU)	1.40	1.55	0.50	1.80	0.85
Max TM Data Rate that can be supported (kbps)	195	150	775	118	500
FULL RATE OBSERVATION @ 75.5 Kbps (No. of days)	30	30	30	30	30
Data Stored On-Board during Full Rate Obs (Gbits)	196	196	196	196	196
LOW RATE OBSERVATION @ 12.5 Kbps (No. of days)	116	108	113	114	104
Data Stored On-Board during Low Rate Obs (Gbits)	125	117	122	123	112
DATA VOLUME GENERATED IN ONE ORBIT (Gbit)	321	312	318	319	308
NO. OF DAYS REQUIRED TO DOWNLOAD THE OVERALL DATA VOLUME AT MAX DATA RATE (8h/day G/S support)	**57**	**72**	**14**	**94**	**21**
Margin (No. of Days)	45	23	88	8	81

18 ANTENNAE

The antenna subsystem is composed of the high gain antenna and the low gain antennas. The low gain antennas must provide Earth coverage during the entire mission. The high gain antenna is used to download the data during the nominal and extended mission phases.

18.1 High Gain Antenna Assembly

18.1.1 Requirements and Design Drivers

The design of the High Gain Antenna must take into account the following requirements:

- High operative temperature and thermal stability
- Dual frequency band (X/Ka)
- The antenna size is constrained by spacecraft accommodation.
- Must provide wide coverage: the steering range must be ±180 deg in Azimuth and 52/-42 deg in Elevation
- Pointing stability (0.1 deg)

18.1.2 Assumptions and Trade-offs

Two trade-offs have been considered, related to the type of antenna and to its position and operation.

Antenna Configuration

As it is a dual band antenna, the following configurations can be considered:

1. Single on-set reflector illuminated by dual-band feed
2. Cassegrain on-set antenna with metallic subreflector illuminated by dual frequency band feed.
3. Cassegrain on-set antenna with dichroic subreflector, transparent at Ka band and reflective at X band.
4. Cassegrain on-set antenna, dichroic subreflector, reflective at Ka band and transparent at X band.

Options 2 and 3 are of interest for thermal reasons, basically because they don’t include any dielectric material. Option 4 is the best option from the RF point of view: the blockage is minor and the performance can be optimised at each band. It has a strong Mercury heritage and the high temperature materials needed, can be assumed to be available from Mercury cornerstone, if the temperature ranges are similar. Therefore, the preferred configuration is option ‘4’, which can be assumed to be the Mercury high gain antenna. Due to the accommodation constraints it has to be a scaled down version of this antenna, i.e. the overall diameter is 1.5 m instead of 1.7 m, which means that the paraboloid reflector diameter is 1.32 m instead of 1.5 m.

Antenna positioning and operation

Assuming that the Mercury antenna is used, two aspects need to be taken into account related to the positioning of the antenna when orbiting around the Sun:

1. From the thermal point of view, the antenna has to be in a similar temperature range to the Mercury mission. This means that it has to be shielded by the Spacecraft body at least when it is close to the Sun.
2. From the communications point of view, the antenna has to point to the Earth during a number of days enough to download the data. Due to the mission profile, if the antenna is shielded by the Spacecraft body, it can only be pointed to the Earth during a limited number of days, not enough to download the data.

The proposed solution is to shield the antenna by the Spacecraft body when the Sun-Spacecraft distance is minor than 0.5 AU. During the rest of the orbit, i.e. when the Sun-Spacecraft distance is greater than 0.5 AU, the antenna is out of the Spacecraft shade, being the nominal position of the boom along the +z axis.

18.1.3 Baseline Design

The proposed antenna is a scaled down version of the Mercury antenna, 1.5m to 1.3m. The antenna is mounted on a 2 axis steerable deployable boom.

Antenna Performance

The approximate antenna performance (see

Table 18-1) has been calculated considering the following optical parameters:

- Circular reflector diameter (D): 1.32 m
- Focal Length (F): 0.55 m
- F/D ratio: 0.42

Band	**Frequency (GHz)**	**Gain (dB)**	**3dB Beamwith (deg)**
X Uplink	7.150	37.3	2.22
X Downlink	8.4	38.7	1.89
Ka	32	50.32	0.5

Table 18-1: High Gain Antenna performance

Operational Modes

As it was said before, the position of the High Gain Antenna and its use vary during the different mission phases:

- Launch: the antenna is folded.
- Cruise and Nominal Observation Mode (Sun-Spacecraft distance smaller than 0.5AU): the antenna is shielded in order to protect it from the Sun. During both phases the antenna is not used.
- Time Share Observation Mode (Sun-Spacecraft distance greater than 0.5AU): the antenna is used to download the data.

See Solar Orbiter Configurations figure in the Configuration chapter.

Radio Blackouts and Occultations

Communications via the High Gain Antenna are constrained by radio blackouts and occultations.

- Radio blackouts occur when the Sun, the Spacecraft and the Earth are aligned, i.e. when the Sun-S/C-Earth angle is bigger or smaller than certain limit angles. These angles are determined by the condition that neither the S/C nor the ground station antenna main beams “see” the Sun and their values are 2.6 deg (Sun between the Spacecraft and Earth) and 179.67 deg (Sun behind the Spacecraft) respectively. There is only a blockage period in orbit #12 between the flight days 2313 and 2324 (14 days) in which communications are not possible.
- Occultations can occur when Venus is between the Spacecraft and the Earth. In our case this situation never happens.

18.1.4 Critical Areas

As can be seen in Table 18-2, the pointing error losses in Ka band are very high, making clear the need of a pointing strategy able to achieve such restrictive requirements. The proposed pointing strategy starts via the LGA link. Due to the wider beamwith at X band, in the next step the HGA is pointed using this frequency band. Finally, the accurate pointing is carried out in Ka band.

A more detailed analysis is required in the following study phases.

Overall pointing error (Deg)	X downlink (8.4 GHz)	Ka downlink (32 GHz)
0.05	0.01 dB	0.11 dB
0.1	0.03 dB	0.44 dB
0.2	0.12 dB	1.84 dB
0.25	0.19 dB	3.0 dB

Table 18-2: Pointing error losses

18.2 Low Gain Antennas Assembly

18.2.1 Requirements and Design Drivers

- During all the mission phases, it must be possible to establish contact with the Spacecraft via the Low Gain Antennas. Therefore, the number of low gain antennas and their position must guarantee a quasi-omnidirectional coverage
- High operative temperature and thermal stability

18.2.2 Assumptions and Trade-offs

The low gain antennas proposed are X band horns similar to those proposed to Mercury. Their mean features are:

- Gain: 9.5 dBi
- 0 dBi gain at 60 deg

The low gain antennas must provide Earth coverage during the entire mission. In order to minimise their number, their orientation has to be optimised. Starting from the Spacecraft attitude data, and assuming that the beamwidth of each low gain antenna is ±60 deg it has been found that with 4 low gain antennas properly oriented a 99.73% time coverage can be achieved.

18.2.3 Baseline Design

The orientation of the 4 Low Gain Antennas and an approximate representation of their coverage can be seen in

Table 18-3 and Figure 18-1 and Figure 18-2 respectively.

	Azimuth, ϕ (deg)	**Elevation, θ (deg)**
LGA 1	0	110
LGA 2	180	95
LGA 3	90	120
LGA 4	270	68

Table 18-3: Orientation of the Low Gain Antennas (in spherical co-ordinates with respect to the satellite axes)

18.2.4 Critical Areas

It has to be noted that in the proposed solution neither the exact position of the antenna, that is constrained by their thermal protection, nor the influence of the Spacecraft (e.g. solar panels) on their radiation pattern have been taken into account. In order to optimise the location of the Low Gain Antennas both problems must be evaluated.

Due to the minor distance to the Sun with respect to the Mercury mission, the thermal protection of the antennas must be guaranteed. If due to their location, they are exposed to the Sun, higher temperature materials development could be needed.

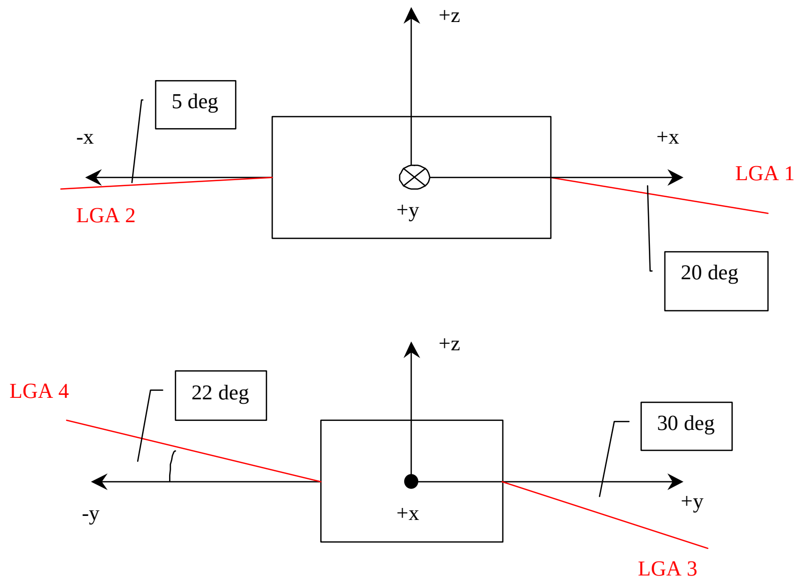


Figure 18-1: Low Gain Antennas orientation

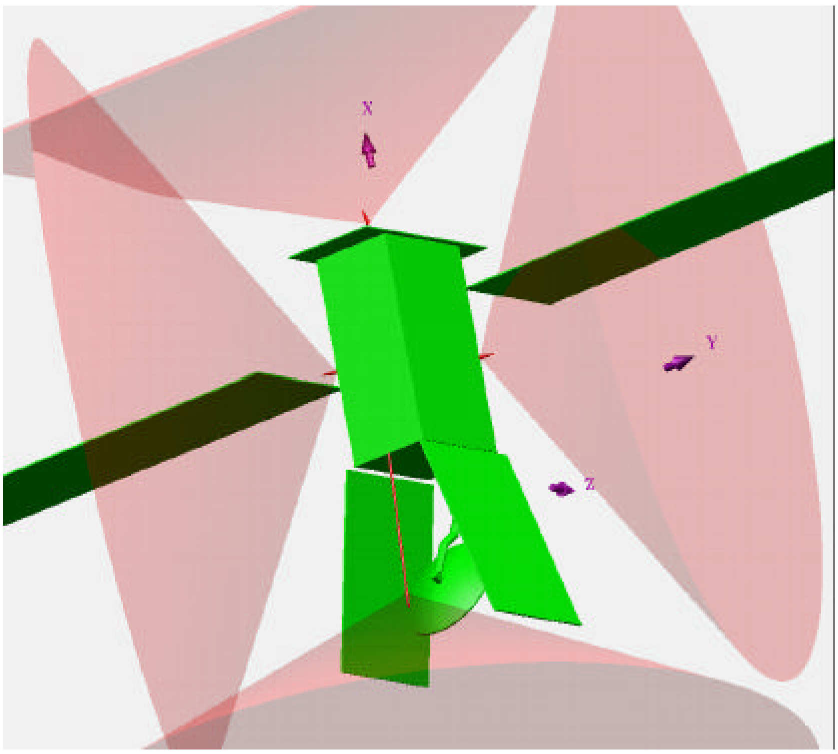


Figure 18-2: Low Gain Antennas positioning and field of view

19 STRUCTURE

19.1 Driving Requirements

19.1.1 Launcher

The launcher taken as a reference is SOYUZ-ST that uses the FREGAT upper stage, but it must be mentioned here that the environment for this launcher is not yet fully defined since the adaptation of the upper stage and of the Ariane 4 long fairing is still in progress. For the purpose of this study, the design requirements defined for SOYUZ have been used; they might need to be updated for SOYUZ-ST.

19.1.2 Design Requirements

Structural Design Requirements are considered typical for this type of mission (following ECSS E-30).

19.2 Structure description

The structure is a parallelepiped shaped body that is split in two modules which provides several benefits:

- high thermo-elastic stability is provided by the payload module which houses the instruments that have high pointing stability requirements
- flexibility in the AIV programme, and possibility to integrate and verify concurrently the two modules. In particular, it allows easy access to the propulsion system, i.e. propellant tank between the PLM central cylinder and the SVM cone and the piping and valves integrated on the SVM upper platform
- cost savings as the service module is constructed with standard aluminium honeycomb panels.

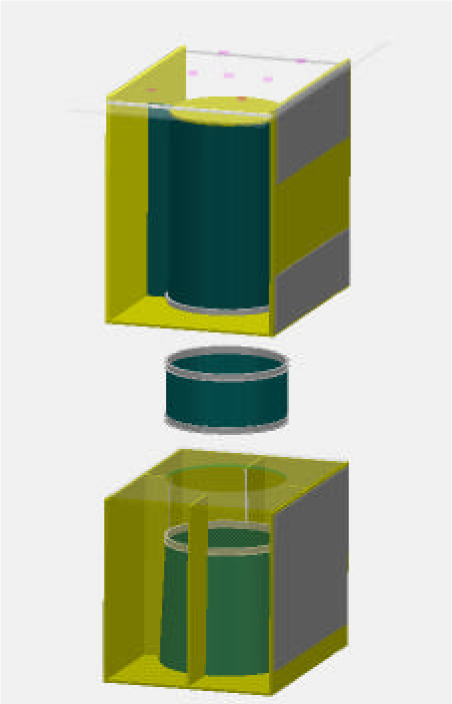


Table 19-1 –Structural Layout

19.2.1 Payload module

It includes a central cylinder that provides the required stiffness. Two shear walls link the central cylinder to the +Y and –Y panels. This results in a lightweight H-shaped beam that provides a very good combination of stiffness and accessibility to the instruments and equipment.

The central cylinder is also connected to the service module with an aluminium interface ring that supports also the –X platform. At its other end is attached a heat-screen made of a lightweight Carbon Fibre Reinforced Plastic (CFRP) panel.

The PLM structure is made of aluminium sandwich construction with CFRP face skins for thermo-elastic stability reasons because it houses the four main instruments. The instruments are iso-statically attached to the central cylinder.

19.2.2 Service module

The service module that does not have the same stringent thermo-elastic stability requirements as the payload module is made of honeycomb sandwich with aluminium facesheets. This is a cost-effective lightweight design.

The cruise solar generator is made of two wings featuring two panels each. The wings are attached to two opposite walls of the satellite body when stowed. In addition, another solar generator is required for the orbital phase. Description of holding points and of the deployment scheme is also addressed in the Mechanisms chapter.

19.2.3 Launcher interface

The FREGAT upper stage features 8 attachment points that are on a 2000-mm diameter circle. Given the size of the lower face of the satellite, it is therefore necessary to use a conical adapter.

A cost-effective solution would be to use the 1194-mm interface ring because Cluster Project has already developed and qualified an adapter for this diameter. However this adapter is heavy (105 kg) and would require to increase the spacecraft cross section with impact on the overall mass and on the thermal input.

The current baseline is the standard 937 mm interface ring that provides adequate connection to the adapter together with significant mass savings on the launch mass (about 5 kg) but this requires to design and qualify a new adapter that uses the upper part of an Ariane adapter. A similar adapter was proposed by industry for Mars Express, using a conventional 937mm Ariane interface ring and the Ariane CFRP adapter technology. This adapter represents significant mass savings (over 50 kg) with respect to the current Russian design. A detailed proposal was made to the Agency including design drawings and an industrial price.

19.3 Structure mass budget

Item	Nr.	M_struct [kg]	Material
TOTAL STRUCTURE		125.73	
Service Module		64.507	
Adapter ring	1	6.930	Aluminum
SVM Cone	1	15.540	CFRP/Honeycomb
SVM+Y panel	1	6.078	Al/Honeycomb
SVM-Y panel	1	6.238	Al/Honeycomb
SVM+Z panel	1	5.418	Al/Honeycomb
SVM-Z panel	1	5.318	Al/Honeycomb
SVM X-Y shear wall	2	3.454	Al/Honeycomb
SVM X-Z shear wall	2	4.568	Al/Honeycomb
SVM-X platform	1	6.029	Al/Honeycomb
SVM+X platform	1	5.949	Al/Honeycomb
SVMPLM I/F ring	1	5.917	Aluminum
Payload Module		61.224	
PLM Cylinder	1	12.969	CFRP/Honeycomb
PLM+Y panel	1	10.397	CFRP/Honeycomb
PLM-Y panel	1	10.357	CFRP/Honeycomb
PLM shear wall	2	5.262	CFRP/Honeycomb
PLM-X platform	1	8.251	CFRP/Honeycomb
PLM Heat-screen	1	8.071	CFRP/Honeycomb
PLMSVM I/F ring	1	5.917	Aluminum

Table 19-3: Structure subsystem mass budget

20 GROUND SYSTEMS AND OPERATIONS

20.1 Ground segment facilities and services

All facilities established for Solar Orbiter will be based on extension of the existing ground segment infrastructure, tailored to meet the requirement of the Solar Orbiter mission.

Ground Operations Facilities consist of:

- The Ground Stations and the Communications Network
- The Mission Control Centre (infrastructure, computer H/W).
- The Flight Control System (Data processing and flight dynamics software).

An overview of the Ground Systems is shown in

Figure 20-1.

20.1.1 Ground Stations and Communications Network

It is assumed that the 35-m ESA Station at Perth will be used for contact with the spacecraft during all mission phases.

For the cruise phase to the Sun , the Perth station can be assumed to be sufficiently available for operations. For the in orbit phase, the assumption on the availability of the Perth Station for at least 8 hours per day is for costing only. Which Station will actually be available has to be clarified at a later time. A possibility to be investigated is the use of the 70 m Station in Sardinia for those times when Perth will not be free.

The planning for the 35-m Station at Perth at the moment does not allow to fully provide the resources (station time) required by Solar Orbiter during the Nominal mission phase, due to use of the station by other missions.

During the first 10 days of the mission and during critical mission phases the ESA 15 m station at Kourou will also be used. The 15 m station at Villafranca will be available as back up.

All ESA Stations will interface to the Control Centre in ESOC Darmstadt via the OPSNET communications network. OPSNET is a closed Wide Area Network for Data (telecommand, telemetry, tracking data, station monitoring and control data) and voice.

20.1.2 Mission Control Centre

The Solar Orbiter will be operated by ESOC, and it will be controlled from the Mission Control Centre (MCC) which consists of the Main Control Room (MCR) augmented by the Flight Dynamics Room (FDR), dedicated control rooms (DCRs) and Project Support Rooms (PSRs). During periods around major mission events mainly during Launch and critical period of the transfer phase (LEOP and TO – first 18-20 months) the MCR will be used for Solar Orbiter mission control. During the science operations phase (nominal and eventually extended) and also during low activities periods in the transfer the mission control will be conducted from a Dedicated Control Room.

The control centre is equipped with workstations giving access to the different computer systems used for different tasks of operational data processing. It will be staffed by

- Dedicated Solar Orbiter spacecraft operations staff
- Experts in S/C control, flight dynamics and network control, available on a shared basis for the full mission duration.

20.1.3 Computer facilities and Network

The computer configuration used in the Mission Control Centre for the Solar Orbiter mission will be derived from existing infrastructure.The computer system basically consists of:

- A mission dedicated computer system (workstation hosting SCOS 2000) used for real time telemetry processing and for command preparation and execution, telemetry and command log archiving, and also for non real time mission planning and mission evaluation.
- Workstation hosting the flight dynamics system (ORATOS)
- Workstation hosting the science telemetry data distribution and instrument command reception system (data servers)
- The simulation computer, providing an image of the S/C system during ground segment verification, for staff training and during operations

All computer systems in the control centre will be redundant with common access to data storage facilities and peripherals. All computing systems will be connected by Local Area Network (LAN) to allow transfer of data at sufficient speed and to allow joint access to data files.

The external connections to the Science Operations Team, Science Data Processing Centres and PI's will use commercial/public networks (Wide Area Network).

20.1.4 The Flight Control Software System

A Flight Control System based on infrastructure development (SCOS 2000), using a distributed H/W and S/W architecture for all spacecraft monitoring and control activities will be established including:

- Telemetry reception facilities for acquisition, quality checking, filing and distribution
- Telemetry analysis facilities for status/limit checking, trend evaluation
- Telecommand processing facilities for the generation of command for control, master schedule updates, and on-board S/W maintenance. Their uplink and verification
- Monitoring of instrument housekeeping telemetry for certain parameters which affect spacecraft safety and command acceptance and execution verification
- Separation and forwarding of payload telemetry to Science Data Processing Centres
- Checking, reformatting, scheduling command request for payload

Within the SCOS 2000 system mission specific S/W development will be developed wherever necessary

20.2 Mission Operations Concept

20.2.1 General

The Solar Orbiter mission operation will comprise:

- Spacecraft Operations, consisting of mission planning spacecraft monitoring and control and all orbit and attitude determination and control
- Science instruments operations, consisting of the implementation of the observation schedules and collection and data quality control of the science telemetry.

Mission Operations proper will commence at separation of the Solar Orbiter S/C from the launcher and will continue until the end of the mission, when ground contact to the spacecraft will be aborted. Mission Operation will comprise the following tasks:

- Mission Planning ,long term planning and short term planning (24 hours to one week time frame)
- Spacecraft status monitoring
- Spacecraft control, based on monitoring and following the Flight Operations Plan and the short-term plan.
- Orbit determination and control using tracking data and implementing orbit manoeuvres
- Attitude determination and control based on the processed attitude sensors data in the spacecraft telemetry and by commanded updates of control parameters in the on-board attitude control system
- On-board S/W maintenance
- Operations support for the experimenters in terms of telemetry packet routing and command checking with respects to Spacecraft safety, and telecommand uplink
- Maintenance of ESA ground facilities and network

20.2.2 Mission Planning, Spacecraft Monitoring and Control

The Operations support activities for Solar Orbiter will be conducted according to the following concept:

- All operations will be conducted by ESOC according to procedures contained in the FOP (Flight Operations Plan)
- Nominal Spacecraft control during the routine mission phase will be ‘off-line’. Only one Ground Station (Perth TBC) will be used. The spacecraft is therefore assumed to provide on-board capabilities that relieve the ground systems from assessing the performance in real time, and conducting corrective actions on short notice in case of on-board anomalies. The contacts between the Mission Control Centre and the Spacecraft, except for collecting payload and housekeeping telemetry, will therefore primarily be used for pre-programming of those autonomous operation functions on the Spacecraft, and for data collection for off-line status assessment. Anomalies will be normally detected with delay.
- All Solar Orbiter operations (both S/C in orbit and during cruise) will be conducted by uplinking of a master schedule of commands for later executions on the S/C. This schedule

will contain all commands necessary to undertake the S/C and experiment operations in a predictable fashion. A limited number of time tag commands will be used for S/C safety operations. The master schedule will be prepared by a dedicated Mission Planning System, using inputs defined by the PIs.

- Also the payloads will be mainly operated by ESOC. The health of the scientific instruments will be permanently monitored and necessary control actions will be taken following the same procedures as for the S/C subsystems. The telemetry data products received from the instruments on-board the orbiter will be monitored for its data quality before it is delivered to the science consortium performing the science data processing
- ESOC will procure, operate and maintain the facilities in the Mission Control Centre for S/C on –board software maintenance. This is assumed to be off the shelf H/W and infrastructure S/W.
- During the LEOP phase 24 hours of TT&C X band operations will be conducted from the ESOC MCR (Main Control Room)
- During the Cruise Phase, there will be a 1 shift operations 8 hours/day 7 days/week from an ESOC DCR (Dedicated Control Room), TT&C will be done in X band, no Science operations are foreseen in the current baseline
- During the orbit phase (nominal and eventually extended), 1 shift operations 8 hours per day 7 days/week will be maintained from ESOC DCR, with TT&C in X band and Science downlink operations in Ka band, 96 days/orbit for about 8 hours per day to meet the scientific requirements (worst case is in orbit 1,105 days of station visibility for Sun-S/C distance > 0.5 AU)

20.2.3 Orbit and Attitude Control

The Flight Dynamics support will consist of:

- Orbit determination: of the cruise/orbiter S/C during all mission phases using two-way range and coherent two way Doppler tracking data from up to three ground stations. Orbit determination includes tracking data pre-processing, the calibration of all electric propulsion and main engine manoeuvres, and the calibration of thrusters used for orbital correction and ground controlled attitude manoeuvres that are not pure torques.
- Transfer orbit electric propulsion thrust steering optimisation: optimisation and implementation of electric propulsion thrusters steering laws. Includes low thrust navigation, this means correction of the future steering law to compensate for random deviations from the planned trajectory.
- Capture manoeuvre optimisation : the sequence of manoeuvres from the hyperbolic arrival at the sun after separation of the SEP stage to the final nominal orbit will be optimised to minimise propellant consumption and taking into account all operational conditions.
- Attitude Control System Monitoring : monitoring and verification of the on-board functions such as star mapper window and sensitivity setting
- Payload nadir pointing and antenna steering: preparation of attitude manoeuvres and antenna steering schedule
- Manoeuvre command generation : preparation of command sequences or input to master schedule updates related to all orbit and attitude manoeuvres
- Manoeuvre monitoring

- Calibration of thrusters and sensors.

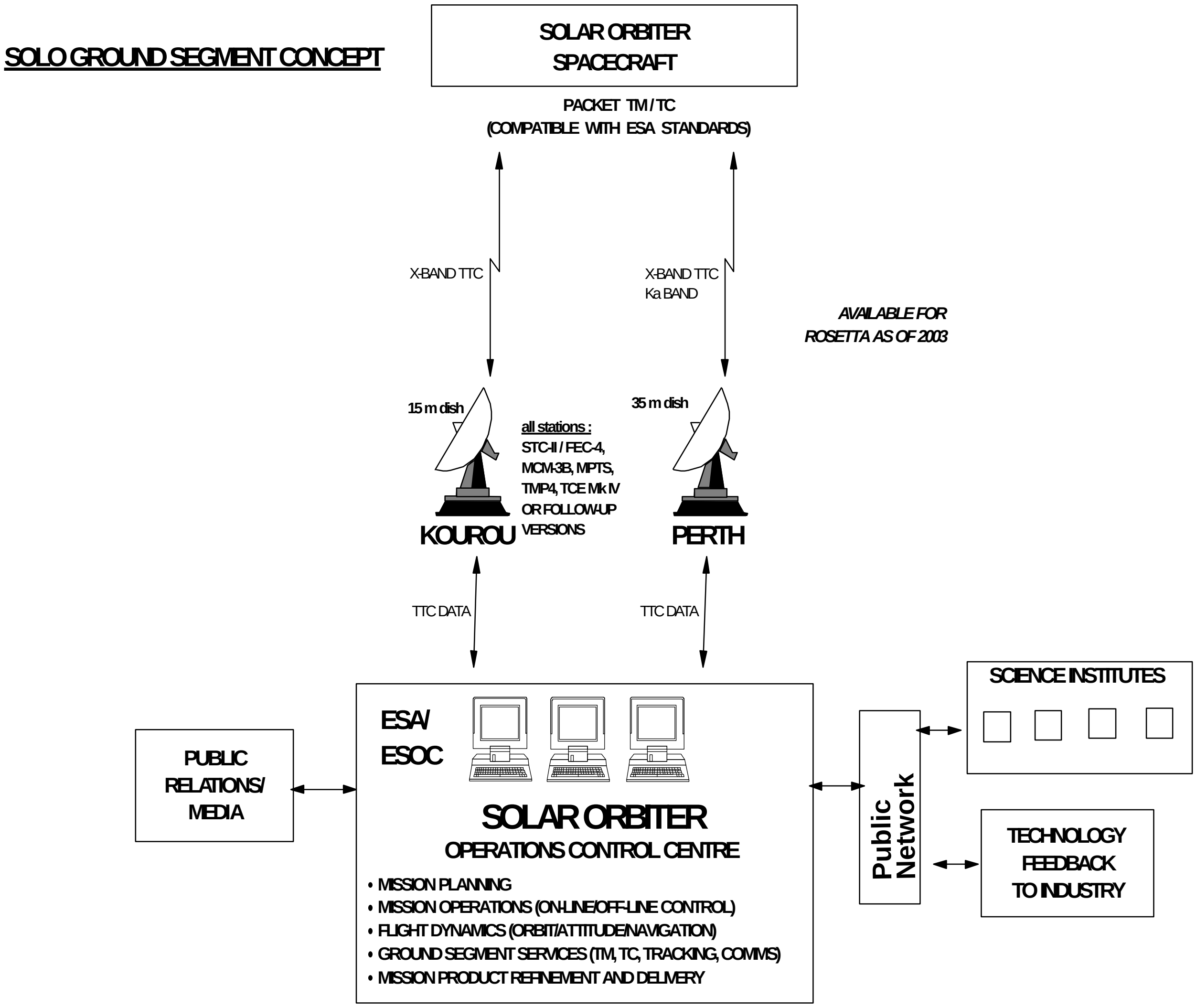


Figure 20-1 –Ground Systems Overview

21 PROGRAMMATICS & COST

21.1 Master schedule

The project Gantt chart below indicates the major phases of the mission development and execution covering the period 2003 to 2015. The schedule foresees the following key phases:

- A development phase covering 2003 to 2008 (phases A to C/D)
- A launch in early 2009
- A cruise phase to achieve solar orbit until mid 2011
- An nominal observation phase lasting 4 years until mid 2015

In addition an extended mission operations phase is foreseen from 20015 until mid 2020. This extended phase is not considered as part of the baseline mission.

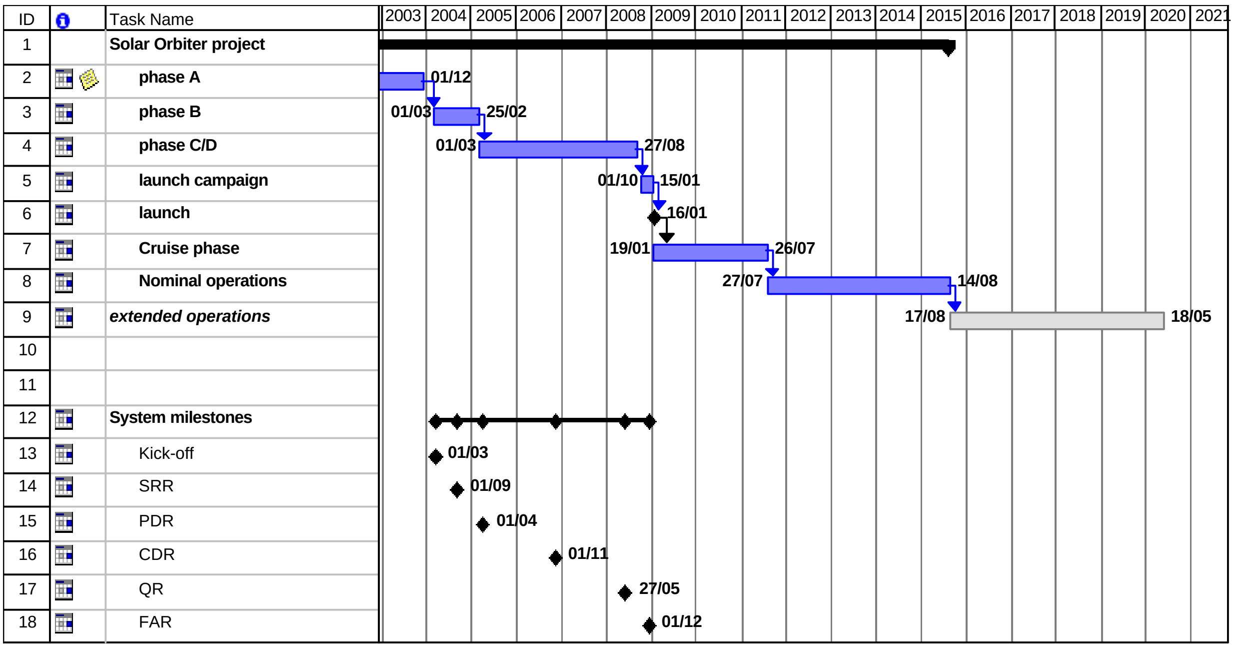


Figure 21-1 –Project Gantt Chart

21.2 Development and model philosophy

The development and AIV philosophy shall be consistent with that of an F class mission. (e.g. MARS-EXPRESS) with the following characteristics:-

- Single contract for phase B/C/D
- Protoflight development programme
- Use of existing equipment from other programmes wherever possible (e.g. Mercury, Mars Express)
- Procure directly at flight standard wherever possible.
- Minimum development of new equipment
- Modified or new equipment built to EM or EQM standard
- Qualification performed on PFM including environmental testing (EMC, mechanical, thermal)

21.3 Special Areas of interest

Due to the nature of the mission, particular attention has to be focused on the thermal qualification/validation of various items requested to operate/survive at 0.21 AU. In this respect, trade-offs of thermal validation with different methods (Analyses, Infrared test, Solar simulation test, etc.) and at different levels (subsystem/equipment) are necessary. The aim of that activity would be to avoid as much as possible any unnecessary test campaigns, in particular at system level, with consequent increase of cost. Furthermore, during trade-off activities, it is necessary to solve the problem associated with the availability of a Sun simulator of ≈ 23 Solar Constants.

The qualification/verification process of the thermal control and of the entire satellite would be strongly influenced by the dimensions (e.g. diameter) of the candidate.

21.4 Main Costing Assumption

The main costing assumption is that Solar Orbiter is considered as a Flexi-mission.

By definition a Flexi-mission is extensively reusing equipment produced for its reference mission; in this case Mercury Cornerstone.

All cost estimates unless specified are therefore taking into account this primary hypothesis.

The main exceptions are the Orbiter Solar Arrays (wings instead of body mounted) and the High Gain Antenna as specified below.

21.4.1 Cost estimate methodology

The following methods have been used in a descending order of priority:

- Reference to similar Off-the-Shelf Equipment foreseen on reference mission.
- Reference to similar Off-the-Shelf Equipment for which we have a reliable price quotation from Industry.
- Reference to similar equipment to be developed.
- Reference to other than above mentioned similar Equipment from the ESA database.
- NB: The differences (Market situation, technical cost drivers…) of similar Equipment mentioned above are studied so to make the necessary adjustments to fit to the Solar Orbiter case.
- Expert judgement from technical specialists when similar references are not available.
- Equipment cost models
- Sub-system cost models
- Ratios for wrap-around costs (like Prime Contractor activities) based on recently observed ratio. (Innovative approaches like Mars Express).

21.4.2 Scope of the cost estimate

According to the study requirements, the full life cycle cost has not been evaluated. Only cost estimates for the space segment are presented. The launcher and the LEOP are excluded.

The cost estimate include:

- A provision for Phase A/B
- The phase C/D costs
- The cost for the launcher adapter ring
- The cost of the propellants

When cost items have been estimated at equipment level, we consider the industrial cost as it would be seen by the prime contractor. It cover the supply of the fight unit(s) with the associated development models when applicable, the spares, the specific GSE and user manual. It also cover for the Project office (i.e.: Management, PA, Engineering) of the equipment supplier.

No margin has been included since it is accounted for at the overall level.

21.4.3 Phase A/B Detailed cost assumptions

Phase A/B has been sized by comparison to Mars Express i.e. a project with a strong prime contractor involvement at the subsystem level of design. No allowance as been made for pre-developments since all technologies are considered to be made available through the reference mission or under technology programs. Planning constraints could imply at the time of Phase B contracting some early starts of Long Lead Equipment suppliers but this amount is then taken into account in the C/D phase estimate.

21.4.4 Phase C/D Detailed cost assumptions

AOCS

The basis of estimate of the AOCS is SMART-1 although its pointing stability is less demanding. It is assumed that existing hardware shall be reused from the reference mission or other sources. No adjustment other than economic conditions has been made.

The Star Tracker is assumed to be supplied by the Danish Technical University (DTU) the same as Mars Express.

Due to its small size, the cost of the RCS propulsion system is calculated using the SSCM98 cost model from the Aerospace Corporation dedicated to small satellites. The resulting calculation has been increased by a complexity factor of 10% for considering solar orbiting instead of earth orbiting. The related non-recurring aspects are deemed being taken into account by the reference mission. The result is consistent with the Mars Express figure of which the main engine is removed and tank greatly reduced, and the addition of small thrusters (from 8 to 12).

Electric Propulsion

The Cost figure is based on the Specialist quotation. Although the figure seems low, the highly competitive market seems able to deliver equipment directly issued from the commercial market for this kind of amount.

Solar Arrays

The Cruiser Solar Arrays are directly derived from the commercial market. It is assumed that the reference mission covers all adaptation costs.

The Orbiter Solar Arrays are based on technologies developed for the reference mission. Some design adaptation is required due to the different configuration (solar panels instead of body mounted). Technology development (cells) is assumed to be derived from the reference mission.

Electrical Power sub-system and Harness

The figures are based on Mars Express cost with adjustments on mass and economic conditions. Only 20% new design is considered compared with the reference mission.

TTC sub-system

The hypothesis for the X/Ka band HGA steerable antenna is a limited design complement w.r.t. the reference mission.

Should the reflector diameter not be the same, it is assumed that some would be available on the market at “recurring price”.

The APM/APE cost is included in the mechanism section.

The dual transponder is a reuse of the adaptation of the SMART-1 Kate experiment to the reference mission.

The power amplifiers (X and Ka) are expected to exist on the market. Limited specific engineering shall be required for adapting to the mission, thanks to the reference mission.

The LGA are directly reused from the reference mission for which they have been designed to cope with the high temperature and radiation environment.

DHS

The DHS has been evaluated independently by the DHS specialist and the cost engineer to arrive to similar (within 10%) figures.

The costs are derived from Mars Express and adapted both with the cost drivers and the technology trends.

Limited redesign has been considered w.r.t. the reference mission. Radiation problems are expected to be solved.

The software is separated from the hardware in a specific cost item together with the AOCS software.

Structure

The cost of structure has been calculated using the small Mission Analysis and Design from J. Wertz and W. Larson Ed 92 calibrated with Mars Express. The CFRP parts cost factor has been taken into account.

Mechanisms

The main mechanisms have been reviewed with the specialist and evaluated individually.

Assumption is made that mechanisms do not see directly the sun and so no extra-cost accounting for extreme temperatures are to be considered.

Jettisoning device is considered as a complex device whose cost must be in the range of the major mechanisms. More precise estimate cannot be done due to the lack of references.

Pyrotechnics

Cost Estimate is based on the number of pyrotechnic devices. The figure is cross-checked with the total satellite cost and in-line with the specialist expectation.

Thermal sub-system

The cost is derived using a detailed calculation based on number of heat pipes, square meters of OSR and MLI, etc…). The thermal sub-system is very sensitive to any design changes in the satellite architecture and will probably be enriched with detailed design considerations at a later stage.

On-Board software

The sizing of software is always very difficult in such early phase when detailed functionality is not yet assessed. The amount is then derived from recent scientific missions like SMART-1.

Payload

Not considered under this estimate. The payload size has however been taken into account to estimate the cost of the Prime Contractor activities. Only payload element integration and testing (AIV) has been calculated.

Prime Contractor's Activities

Prime activities are calculated from ratios related to Total HW/SW costs. The ratio is primarily issued from feedback on recent science project like Mars Express and also SMART-1.

Among others, the prime activities include:

- the design at system, but also at sub-system level,
- the full responsibility for Equipment procurement,
- the direct interaction with the PIs for routine technical matters.

Launcher Adapter Ring

Cost estimate is based on the SMART-1 ALVA price (Alternative Launch Vehicle Adapter) for SOYUZ.

Propellants

Cost per Kg of Xenon supply as been directly used due to the large quantity considered. This amount is very much dependant on Xenon market price at the time of procurement.

The cost for hydrazine is made of a fixed part for container and transport and cost per kg for the hydrazine itself.

It should be noted that the cost of the small quantity of hydrazine is negligible compared with the cost of the Xenon.

21.5 Qualitative Cost Assessment

A qualitative cost assessment is given in the following table, where the estimated cost of the Solar Orbiter mission is compared to Mars Express mission (Flexible mission) and to Mercury (Cornerstone mission). To be sure that the Solar Orbiter mission cost is compatible with a Flexible Mission Budget the sign under the Mars Express column should be either “equal”, i.e. ~, or “less than”, e.g. <. Any other sign means higher cost figure. The fonts in bold indicate the most significant areas of extra cost.

	Mars Express	**Mercury**
• AOCS	>	~
• RCS	~	<
• **Power**		
• Control & Distribution	>	<
• Cruise SA	>	<
• Orbit SA	>	<
• **Communications**		
• TT&C (Telecom. & HK)	>	~
• HGA	>	<
• Thermal Control	>	~
• Structure, adapter	>/~	<
• **(Electrical) Propulsion**	>	<
• On-board S/W	>/~	<
• Ground Operations	~	<
• Launcher	~	<<<
• AIV & Testing	>	<

Table 21-1: Cost Comparison with Mars Express and Mercury.

22 TECHNICAL RISK ASSESSMENT

22.1 General

A preliminary top-level technical Risk Analysis of the Solar Orbiter Spacecraft has been performed. The information used is preliminary and contains large uncertainties. The results are indicative rather than absolute. The risk assessment was restricted to a simple risk model of the system design without consideration of the instruments, the launch vehicle, ground operation and physical and functional propagation phenomena. A very simplified version of the ESA risk assessment method was used and the results were calculated using the ESA risk analysis software RISAN.

The present analysis contains:

- Preliminary assessment of the technical risk, defined as the system unreliability i.e. 1 (System Reliability)
- Preliminary ranking of subsystems according to their technical risk contribution (%) during the Cruise Phase
- Preliminary ranking of subsystems according to their technical risk contribution (%) during the Observation Phase

22.2 Approach

The approach used included:

- Utilisation of reliability data derived from comparison with previous space projects (Artemis, Rosetta, CESAR, Mercury Cornerstone and Mars Express) and individual expert judgement.
- Independence of subsystems.
- Separate analysis for Cruise phase and for Observation phases, in order to see the different ranking of the risk contributors due to the different environment during each phase.
- Consideration of redundancy when applied.
- Consideration of the duty cycles (time on/off) of each subsystem during the Cruise and Observation phases.
- Consideration of radiation during the Observation phase as a possible common cause failure for electronic equipment.

22.3 Preliminary technical risk results.

It is strongly emphasised that the preliminary risk results are an indication and must not be interpreted as absolute numbers.

The indicated risk at system level is as follows:

Technical Risk (probability of Space craft failure per mission)	~0.1

of which the risk is ten times greater in the Observation Phases than in the Cruise Phase.

This high technical risk is in agreement with the results obtained for similar projects.

The percentage contribution (%) to the above technical risk is ranked according to subsystems in the following tables:

Rank	Sub-System	Percentage
1	AOCS	~70%
2	Data Handling	~15%
3	Propulsion, Communications, Mechanisms	~10%
4	Power, Thermal Control, Pyros and Structures	~5%

Table 22-1: Ranking of subsystems Risk contribution during the Cruise Phase.

Rank	Sub-System	Percentage
1	Data Handling	~30%
2	Mechanisms	~30%
3	AOCS	~20%
4	Comms, Power, Thermal Control, Pyros and Structures	~10%

Table 22-2: Ranking of subsystems Risk contribution during the Observation Phases.

The results indicate the main areas of risk at this very early stage of the Solar Orbiter project. For both phases the results are directly related to the duty cycle and the estimated probability of loss of mission due to any particular subsystem. The risk during Cruise phase is dominated by the relatively higher failure rate of gyros in comparison to the other equipment. During the Observation phases the risk to Data Handling dominates due to the increased operating time coupled with the susceptibility of the digital electronic equipment to the increased radiation which increases the risk of failure by an order of magnitude.

22.4 Conclusions

The results of the preliminary technical risk assessment indicate where the first risk reduction efforts should be made.

Large uncertainties exist in the failure-rate data at subsystem level during the Observation Phase due to the lack of information available and the preliminary definition of the design. This is expected to improve with the availability of more information and optimisation of the spacecraft design, particularly with reference to the balance between shielding and the radiation hardening of individual components.

23 SIMULATION AND VISUALISATION

A Solar Orbiter prototype Project TestBed has been developed in the frame of the Solar Orbiter.

The PTB implementation consists mainly of a mission simulator containing functional models of the spacecraft, its subsystems (such as sensors, actuators, antennas, onboard computer, instruments, etc), the spacecraft orbital environment and the ground segment.

The simulation has taken as input the data produced by Mission Analysis. This data consists of:

- satellite position over time
- thrusting periods
- thrust level and direction
- Solar latitude and longitude of both the spacecraft and the Earth

The graphical model of the Solar Orbiter satellite has been taken from the Configuration model and has been integrated in the 3D visualisation. The data has been automatically exported from the CAD model into the VRML modelling language and then imported in the simulation's graphical drawing package.

23.1 Solar Orbiter Simulation Results

During the Solar Orbiter study the PTB has been mainly used for performing the following analyses:

- Communication antennas coverage maximisation and location optimisation (plots)
- Identification of Spacecraft and Earth foot-prints on the Sun surface during science observation periods per orbit (2D map)
- Visualisation and verification of mission phases and operations (3D visualisation)

The use of the simulation confirmed the engineering analysis, as shown in the following sections.

23.1.1 Ground station visibility and antennas angles

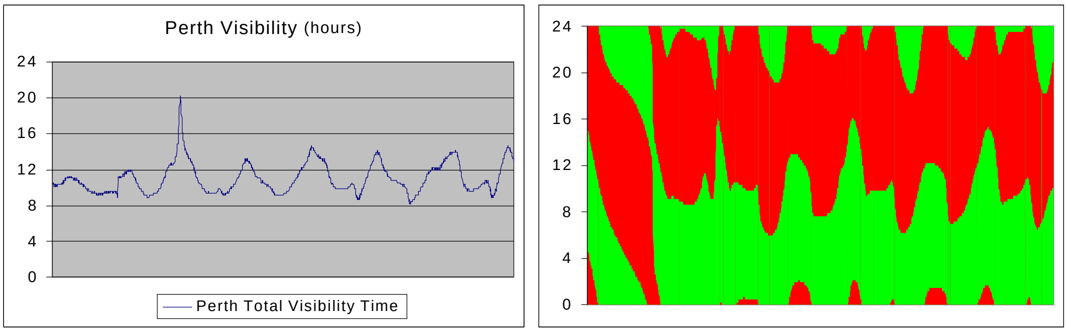


Figure 23-1: Total Visibility Time and Visibility Periods (green) for ground station Perth, with a minimum elevation angle of 5 degrees (downlink constraint), during the complete mission time.

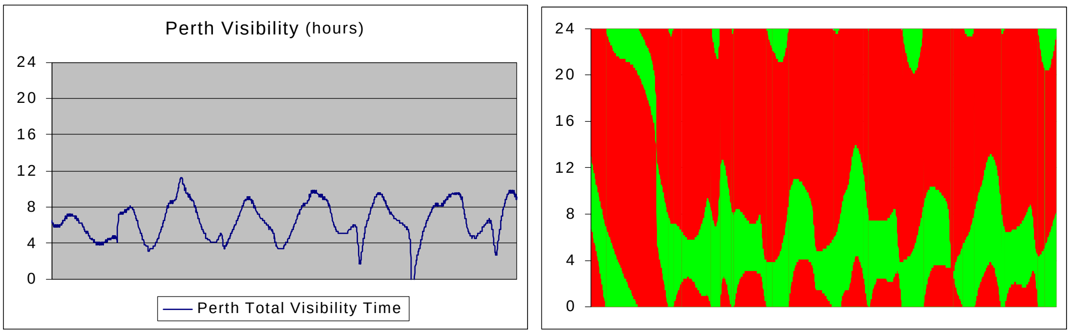


Figure 23-2: Total Visibility Time and Visibility Periods (green) for ground station Perth, with a minimum elevation angle of 30 degrees (uplink constraint), during the complete mission time.

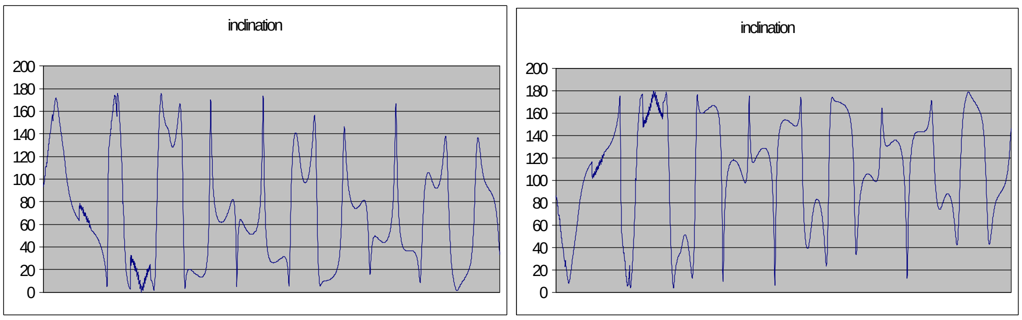


Figure 23-3: Position of the Perth ground station as seen from the Low-Gain Antennas (i.e. the +X and the -X antennas respectively, these value are the inclination angles w.r.t. to the antenna pointing direction in degrees).

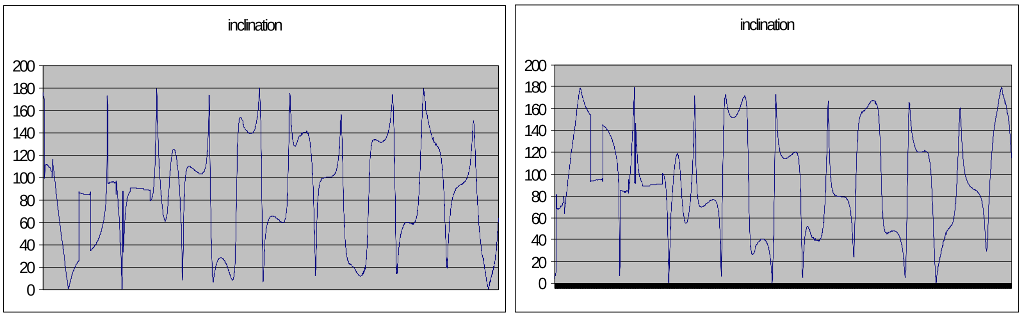


Figure 23-4: Position of the Perth ground station as seen from the Low-Gain Antennas (i.e. the +Y and the –Y antennas respectively, these value are the inclination angles w.r.t. to the antenna pointing direction in degrees).

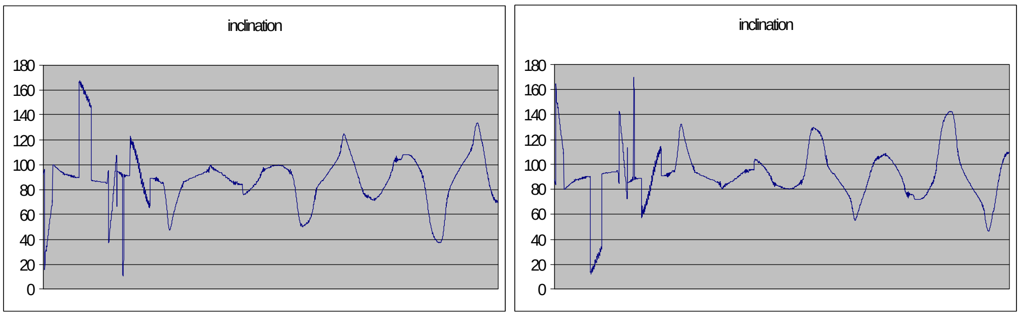


Figure 23-5: Position of the Perth ground station as seen from the Low-Gain Antennas (i.e. the +Z and the –Z antennas respectively, these value are the inclination angles w.r.t. to the antenna pointing direction in degrees).

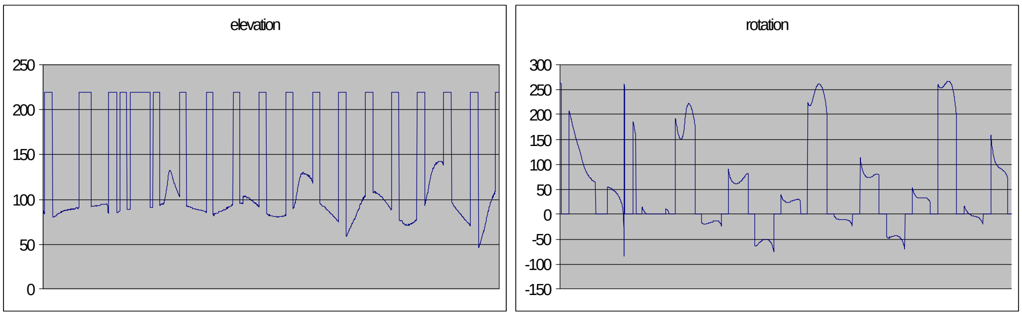


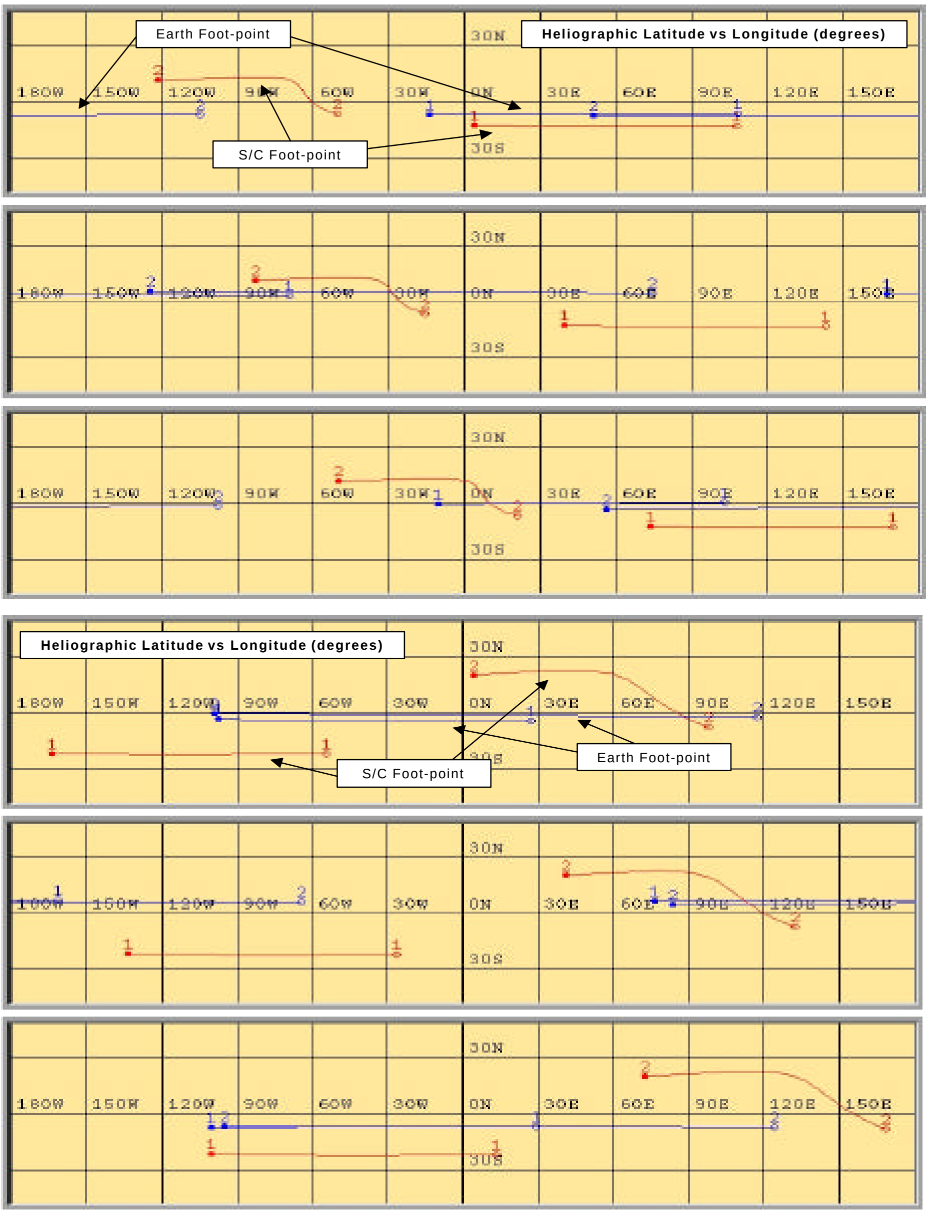
Figure 23-6: Elevation angle and Rotation of the High-Gain Antenna mechanism over the mission time.

23.1.2 Satellite and earth foot-print on sun surface

The following four groups of plots (2D sun maps) provide: Spherical projection of the Sun surface (latitude versus longitude), Spacecraft foot-print on the Sun, Earth foot-print on the Sun, Numbering of foot-print paths according to observation sequence

These plots are made using a spherical projection of the Sun, with the Sun's longitude on the horizontal axis and the Sun's latitude on the vertical axis. Each plot contains three orbits of the satellite around the Sun. After three orbits a Venus swing-by alters the orbit resulting in a higher inclination (so a larger latitude value in the plot). Per orbit, there are three periods of science observations, which are numbered from 1 to 3. Some of the science observation periods overlap, resulting in only two but larger periods.

Orbit 1, 2, 3, Orbit 4, 5, 6



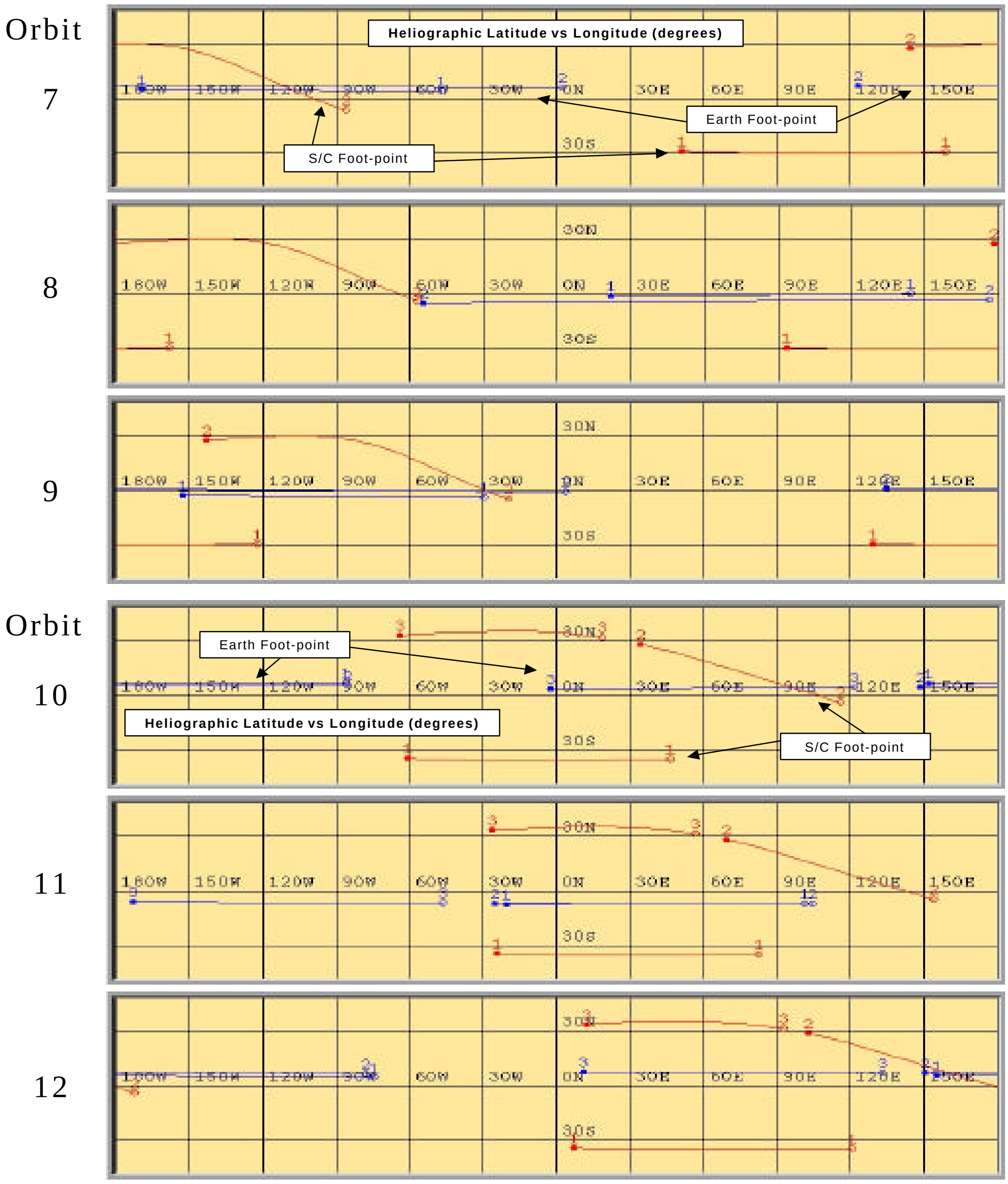


Figure 23-7: Spacecraft Footprint on the Sun over mission orbits.

23.1.3 Solar Orbiter trajectory visualisation

The figure below shows the 3D graphic visualisation of the Solar Orbiter spacecraft trajectory. The satellite trajectory is plotted as a line, using different colours to indicate the various phases of the mission. The meaning of the various colours is listed below.

- Green: cruise phase without thrusting.
- Purple: cruise phase with thrusting.
- Blue: science observation phase.
- Yellow: science observation phase with instruments on (at minimum and maximum solar latitude and at the perihelion passage).
- Red: overlap between minimum/maximum solar latitude and perihelion passage.

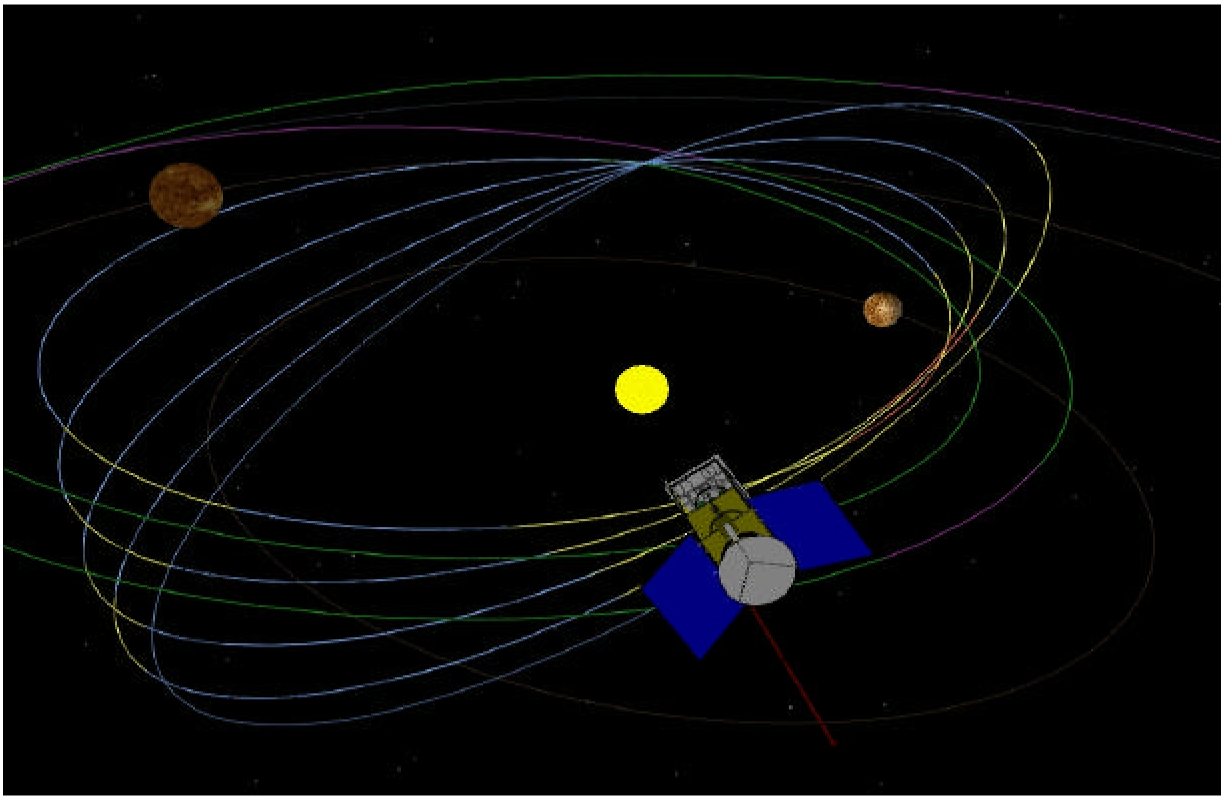


Figure 23-8: 3D Orbit Visualisation using PTB

The 3D visualisation provides the following information:

- Geometric model of the spacecraft imported from Configuration
- Display of spacecraft model with correct position and attitude
- Display of the big solar panels status (available or jettisoned)
- Display of the small solar panels status (reduction of angles if required)
- Display of the high-gain antenna status (position of mechanism, hiding when required)
- Display of high-datarate downlink to Perth
- Display of s/c trajectory from Mission
- Display of textured Earth, Sun, Mercury and Venus
- Display of planet orbits
- Display of stars

24 Final Considerations

24.1 Satisfaction of the Scientific Requirements

A summary analysis of the satisfaction of the scientific requirements by the selected baseline design has been performed, without entering into the merits of each scientific requirement.

The results are as following:

	YES	NO	COMMENTS
1) Orbit			
* Inclination >40°		X	38° latitude @ end extended
* Perihelion <0.3 AU	X		0.21 AU min
* Aphelion at 0.72 AU	X		
* Spiral down		X	Not feasible: too high DeltaV
* Co-rotation	X		10-days /orbit pseudo-co-rotation
2) Launch Date	X		Jan 2009 optimum window
3) Lifetime			
* Cruise ~2 years	X		1.86 y
* Nominal ops ~2 years			2.88 y
* Extended ops ~2 years			2.28 y
4) Sun Pointing			
* 3-axis	X		
* Point.stability (1arcs/15min)	X		3 arcs/15min. Refinement possible only
* Pointing accuracy	X		in Phase B/CD. Too heavy requirement
			at S/C level (high cost impact)
5) Payload Resources			
* Mass 145 kg	X		Kept in present baseline.
* Power 127 W	X		To be optimised later on
* Data rate 74.5 Kbps	X		
* Seven Instrument Packages	X		Radio Experiment out (not defined)

Table 24-1: Summary of Fulfilment of Requirements.

24.2 Areas of further refinement/optimisation

Design areas needed to be refined/optimised in the future:

- thermal control (e.g. heat shield)
- launcher performances (e.g. lift-off mass, launch pad, fairing)
- spacecraft accommodation (e.g.LGAs, HGA antenna)
- communications (e.g.number of LGAs)
- on-ground verification (availability of 22.6 solar constants for testing)
- Solar simulator
- ground station availability (35m antenna)

24.3 Final Considerations

The mission appears to be technically infeasible, based on the assumed programmatic and technical requirements, derived from the scientific requirements, with few areas needing to be refined/optimised.

The mission appears unfeasible within the typical Flexible Mission Budget of 175 MEuro, even after a cost optimisation process. It is very unlikely that the delta cost could be eliminated.

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26 Acronyms and Abbreviations

AGC	Automatic Gain Control
AIU	Attitude and orbit control system Interface Unit
AIV	Assembly, Integration and Verification
Al	Aluminium
AOCS	Attitude and Orbit Control System
APM	mechanisms
ASAP	Ariane Structure for Auxiliary Payload
ASI	Agenzia Spaziale Italiana
AU	Astonomitcal Unit
BDR	Battery Discharge Regulator
BOL	Beginning of Life
BPSK	Binary Phase Shift Keying
BTR	Battery
CCD	Charge Coupled Device
CCSDS	Consultative Committee for Space Data Systems
CDF	Concurrent Design Facility
CDMS	Central Data Management System
CDMU	Central Data Management Unit
CDR	Critical Design Review
CFRP	Carbon Fiber Reinforced Plastic
CME	Coronal Mass Ejections
CMO	Cover slide (SA)
COG	Centre of Gravity
COTS	Commercial Off The Shelf
CP	Commissioning Phase
DEG	Degrees
DHS	Data Handling System
DOD	Depth Of Discharge
DOF	Degrees of Freedom
DPU	Data Processing Unit
Dpx	Diplexer
DTU	Danish Technical University
ECSS	European Co-operation for Space Standardisation
EGSC	Electrical Ground Support Equipment
EIRP	Effective Isotropic Radiated Power
EM	Engineering Model
EOL	End of Life
EQM	Electrical Qualification Model
ESA	European Space Agency
ESOC	European Space Operations Centre
EUV	Extreme Ultra Violet
FAR	Flight Acceptance Review
FC	Flow Control (unit)
FDIR	Failure Detection Isolation and Recovery

FEEP	Field Emission Electric Propulsion
FOP	Flight Operations Plan
FOV	Field Of View
G/S	Ground System
G/T	Gain to Noise temperature ratio
GaAs	Gallium Arsenide
GCR	radiation
GEO	Geostationary Orbit
HGA	High Gain Antenna
H/W	Hardware
HF	High Frequency
HK	House Keeping
HRMS	Hold-down and Release Mechanisms
HST	Hubble Space Telescope
I/F	Interface
IMU	Inertial Measurement Unit
I/O	Input - Output
ITU	Intelligent Terminal Unit
LCL	power
LEO	Low Earth Orbit
LEOP	Launch Early Orbit Phase
LF	Low Frequency
LGA	Low Gain Antenna
LOS	Line Of Sight
LU	Latch Ups
LV	Launch Vehicle
MCC	Main Control Centre
MCR	Main Control Room
MDS	Mission Dissemination System
MGSC	Mechanical Ground Support Equipment
MIL	Military (Standard)
MIP	Mission Implementation Plan
MLI	Multi-Layer Insulation
MM	Mass Memory
MMS	Matra Marconi Space
MOI	Moment of Inertia
MPPT	Maximum Power Point Tracking
MPS	Mission Planning System
MT	Magnetic Torquers
NA	Not Applicable
NASA	National Aeronautics and Space Administration
NIEL	Non-Ionising Energy Loss
OCC	Operations Control Centre
OP	Operational Phase
Ops	Operations
OSR	Optical Surface Reflector

OTS	Off-The-Shelf
P/L	Payload
PA	Product Assurance
PCM	Pulse Code Modulation
PCU	Power Conditioning Unit
PDR	Preliminary Design Review
PDU	Power Distribution Unit
PFM	Proto-Flight Model
PI	Principle Investigator
PLM	Payload Module
PM	Phase Modulation
PMM	Power Maximum
PPU	Power Processing Unit
PR	Pressure Regulator
PSR	Project Support Room
PTB	Project Test Bed
PWP	Plasma Wave Package
QA	Quality Assurance
QR	Qualification Review
RCS	Reaction Control System
RF	Radio Frequency
RFDU	Radio Frequency Distribution Unit
ROM	Rough Order of Magnitude
RPM	Revolutions Per Minute
RTU	Remote Terminal Unit
RW	Reaction Wheel
Rx	Receiver
S/C	Spacecraft
S/S	Sub-System
SA	Solar Array
SAD	Solar Array Drives
SAS	Sun Acquisition Sensor
SCA	Solar Arrays
SCOS	Spacecraft Operation System
SEP	Solar Electric Propulsion
SEU	Single Event Upset
Si	Silicon
SIL	Space Innovations Limited
SM	Switch Matrix
SOHO	Solar and Heliospheric Observatory
Specs	Specifications
SPPG	Solar Physics Planning Group
SPT	Stationary Plasma Thruster
SR	Solar Radii
SRR	System Requirements Review
SSM	Second Surface Mirror

SSO	Sun Synchronous Orbit
STM	Structural and Thermal Model
STR	Star Tracker
SVM	Service Module
SVT	System Validation Test
SW	Software
TBC	To Be Confirmed
TBD	To Be Determined
TC	Tele-Command
TCU	Thermal Control Unit
TM	Telemetry
TRACE	Transition Region and Coronal Explorer
TT&C	Telemetry, Tracking and Command
Tx	Transmitter
WRT	With Respect To

27 Appendix –Analysis of Radiation Environment

27.1 Introduction

The Solar Orbiter mission evaluated consists of three phases: cruise, operational and extended mission, with a launch date in 2009.

	Cruise Phase	Operational Phase	Extended Phase
Average Periapsis	0.437 AU	0.229 AU	0.306 AU
Average Apoapsis	1.038 AU	0.875 AU	0.798 AU
Average distance Sun-S/C	0.668 AU	0.447 AU	0.494 AU
Average Inclination	1.728°	13.069°	27.017°
Duration	1.86 yr.	2.88 yr.	2.28 yr.

Table 27-1: Orbital Parameters for Radiation Analysis

27.2 Radiation Environment

The space radiation environment presents a major problem to space systems. The environment consists of geomagnetically trapped charged particles, solar protons and galactic cosmic rays. It is the penetrating particles that provide the main problems, which include upsets to electronics, payload interference, damage to components and deep dielectric charging.

Solar protons are products of solar events, with energies in excess of several hundred MeV and peak fluxes in excess of 10^6 Protons/cm^2/sec for protons with energies greater than 10 MeV. These events, though, are relatively rare, occurring primarily during periods of solar maximum activity, which commences 2.5 years prior to sun spot maximum and lasting for seven years. The duration of individual events is usually on the order of days. The large fluxes of energetic protons and heavier ions can contribute a large dose, increase upset rates in electronics and increase radiation induced background noise in detectors.

The solar proton fluence model used in this analysis was that developed by Feynman et al. at JPL [Ref.3]. This model uses a data set spanning three solar cycles. Spacecraft engineers are replacing the older King model [Ref.4] as the standard solar proton model with the JPL model. The interplanetary nature of the mission removes the spacecraft from the effective region for geomagnetic shielding; therefore, no geomagnetic shielding attenuation of the solar proton fluence spectra was performed. The JPL model, though, was developed for proton fluences in Earth orbit; further scaling of the fluences for heliocentric distance must be performed. The inverse square of the time average heliocentric distance during the phase is used as the scaling factor. For a worst case scenario, an inverse cube can be employed assuming a large event, c.f. August '72, occurs at perihelion.

27.3 Radiation Effects

The primary effects of radiation are due to ionising dose, non-ionising dose (bulk damage displacement), solar cell degradation, and single event upsets/latch-ups (SEU/LU) in components.

The solar proton spectra are used to calculate the ionising dose deposition in a minuscule silicon target as a function of spherical aluminium shell shielding thickness using the SHIELDOSE [Ref. 5] code. This simplified particle transport code is well suited for routine dose predictions in situations where the geometrical and compositional complexities of the spacecraft are not well known.

The non-ionising dose-depth curve is similarly calculated for a minuscule silicon target in a spherical shell of aluminium using the Non-Ionising Energy Loss curve (NIEL). It is primarily optical components, such as opto-couplers and CCDs, which suffer degradation from non-ionising dose. Whereas the ionising dose is cause by a variety of particle species (gamma, electron, proton and ion), the non-ionising dose derives from protons and ion fluxes. Component testing performed with a Cobalt-60 source (gamma ray) will not be applicable for assessing the non-ionising dose degradation.

The solar proton spectra are also used to calculate the solar cell damage equivalent fluence of 1 MeV electrons as a function of cover glass thickness with the EQFRUX silicon solar cell code [Ref. 5]. Infinite cell back shielding is assumed and 10 MeV proton to 1 MeV electron equivalence ratios of 3000 for the silicon solar cells. For body mounted solar cells, the infinite back shielding condition applies. However, for "wing" type solar arrays with thick cover glass, rear incidence irradiation should be considered. A lightweight panel construction, e.g. as used on HST, could result in a significant radiation contribution caused by a lack of adequate shielding on the array's backside.

27.4 Results

27.4.1 Cruise Phase - 1.86 yr.

The solar proton fluence spectrum for the cruise phase was calculated based on a 95% confidence that the fluence would not be exceeded, an inverse squared scaling factor of 2.24, and a duration of 1.86 years.

27.4.2 Operational Phase - 2.88 yr.

The solar proton fluence spectrum for the operational phase was calculated based on a 90% confidence level, an inverse squared scaling factor of 5.00, and a duration of 2.88 years.

27.4.3 Extended Phase - 5.16 yr.

The solar proton fluence spectrum for the extended phase was calculated based on a 90% confidence level, an inverse squared scaling factor of 4.56, and a duration of 5.16 years. It was assumed that the extended phase would be a continuation of the operational phase, and so the fluence spectrum representative of the operational and extended phases is presented.

27.4.4 Total Mission - 7.02 yr.

The solar proton fluence spectrum for the entire mission was calculated based on a 90% confidence level, an inverse squared scaling factor of 3.70 and a duration of 7.02 years.

27.4.5 Figures

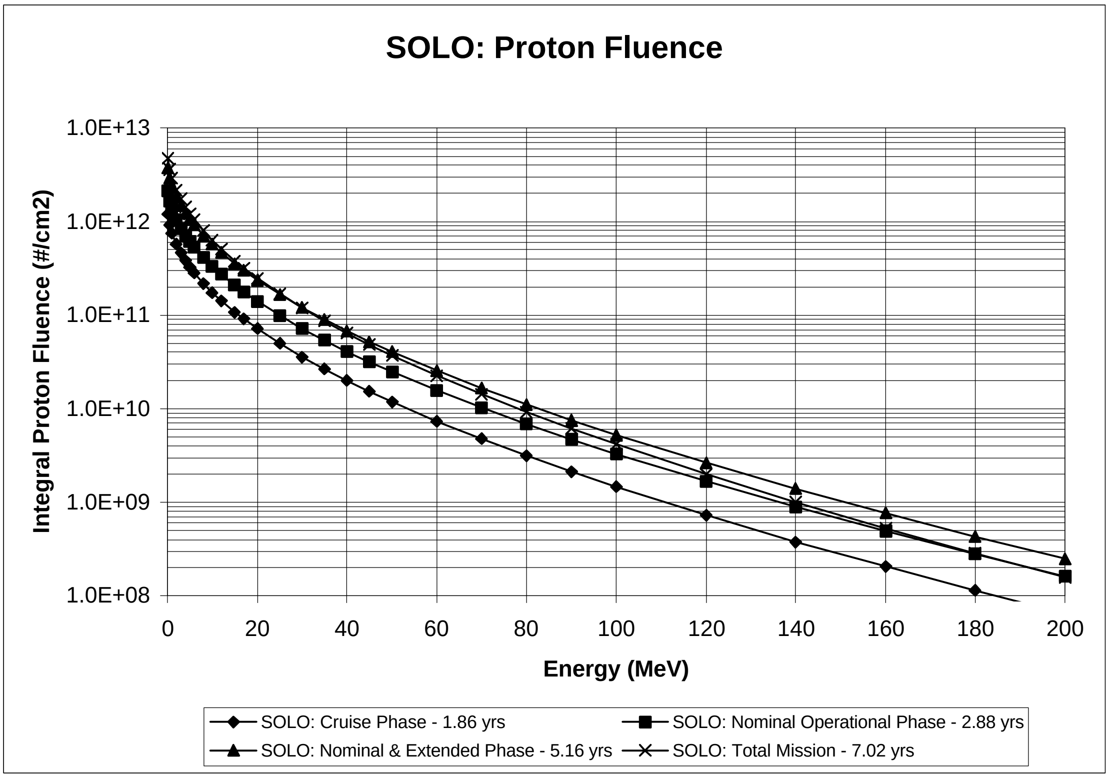

Figure 27-1: Solar Orbiter Proton Fluence

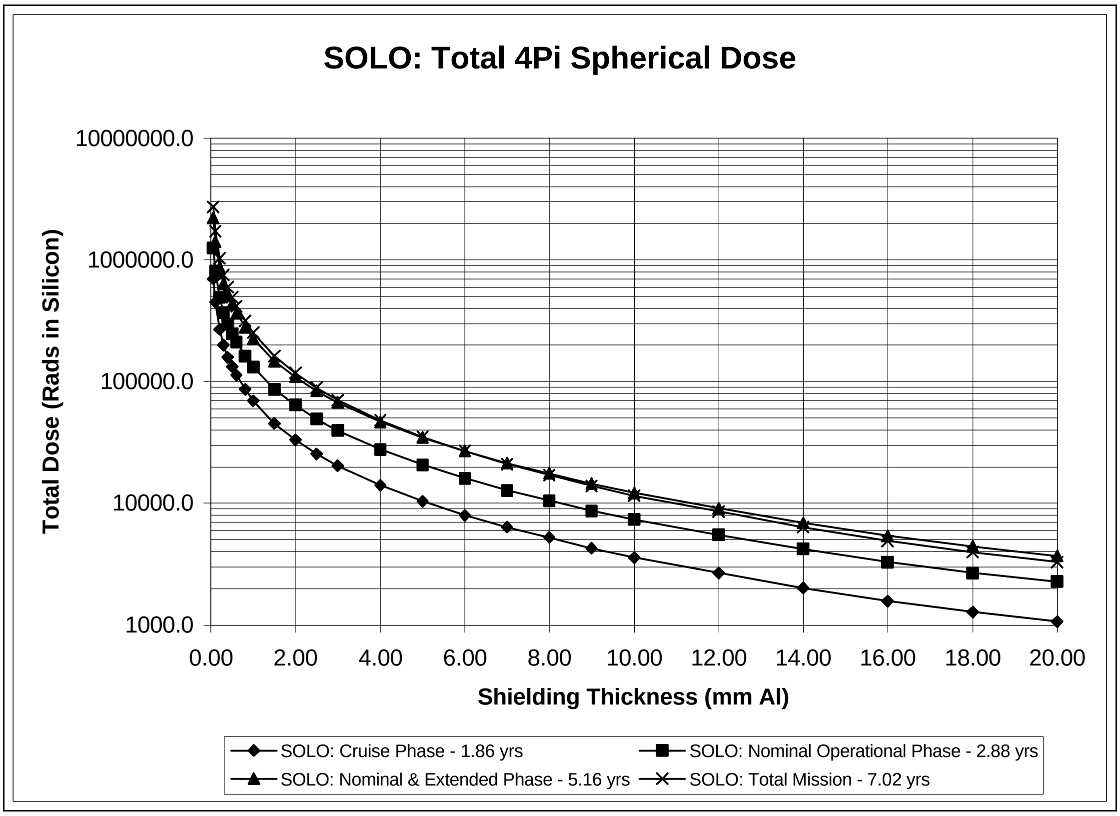

Figure 27-2: Solar Orbiter Total 4Pi Spherical Dose

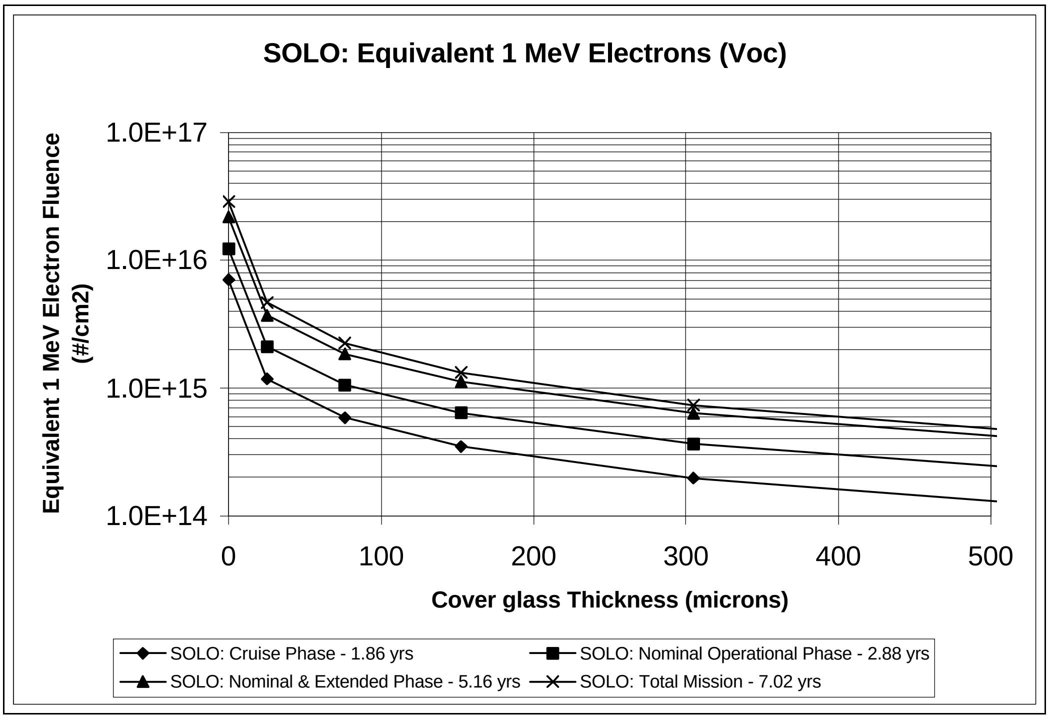

Figure 27-3: Solar Orbiter Equivalent 1 MeV Electrons (Voc)

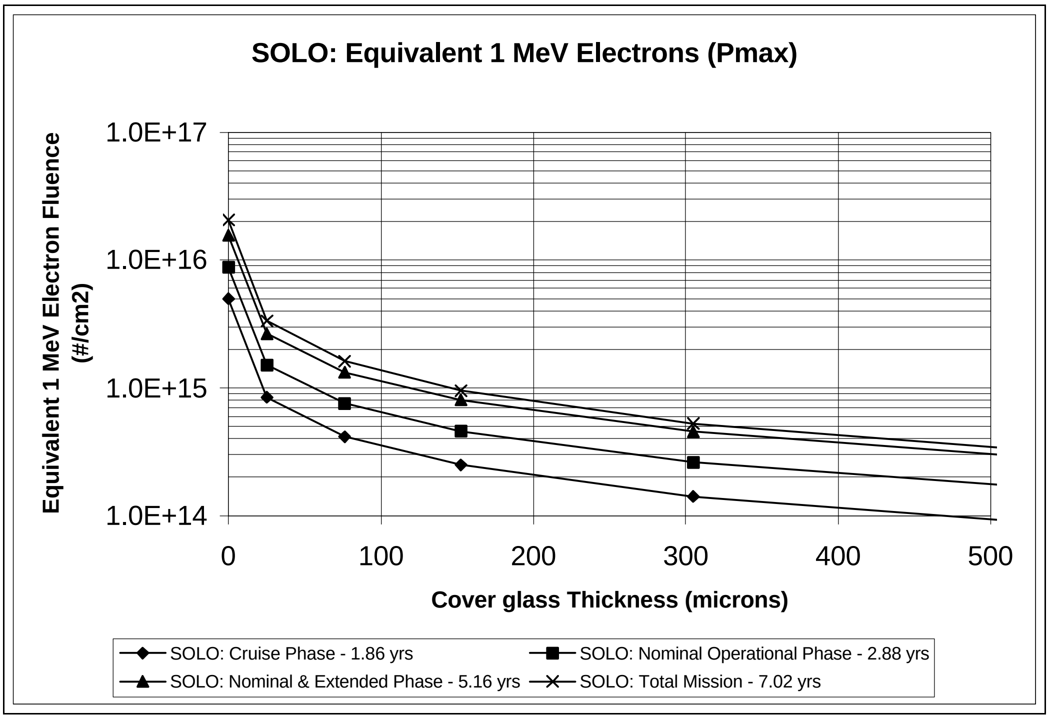

Figure 27-4: Solar Orbiter Equivalent 1MeV Electrons (Pmax)

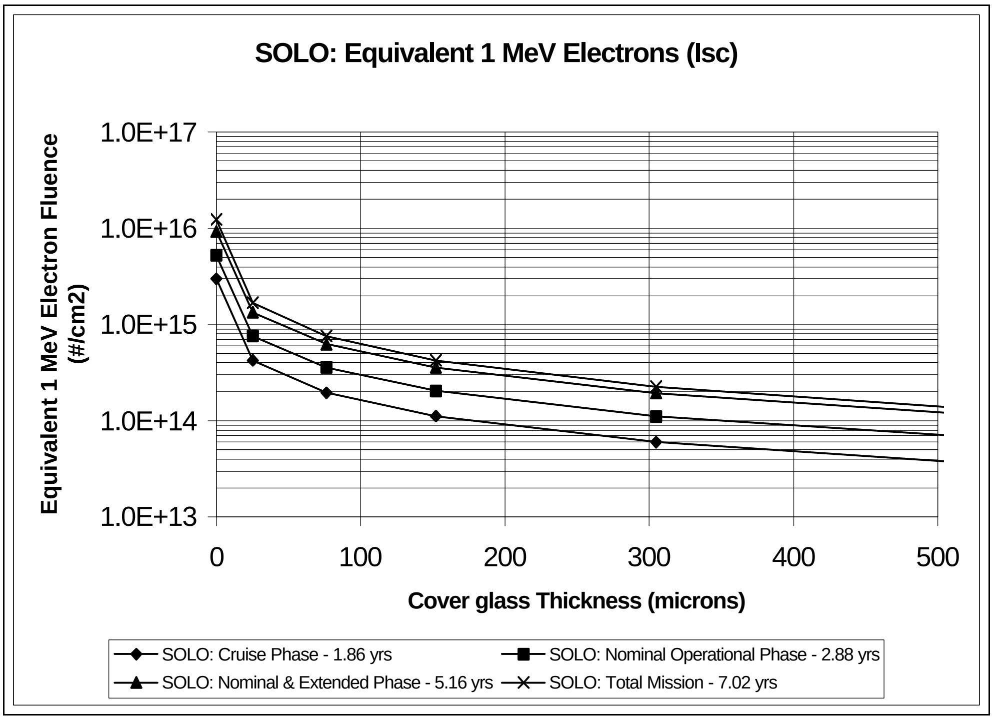

Figure 27-5: Solar Orbiter Equivalent 1 MeV Electrons (Isc)

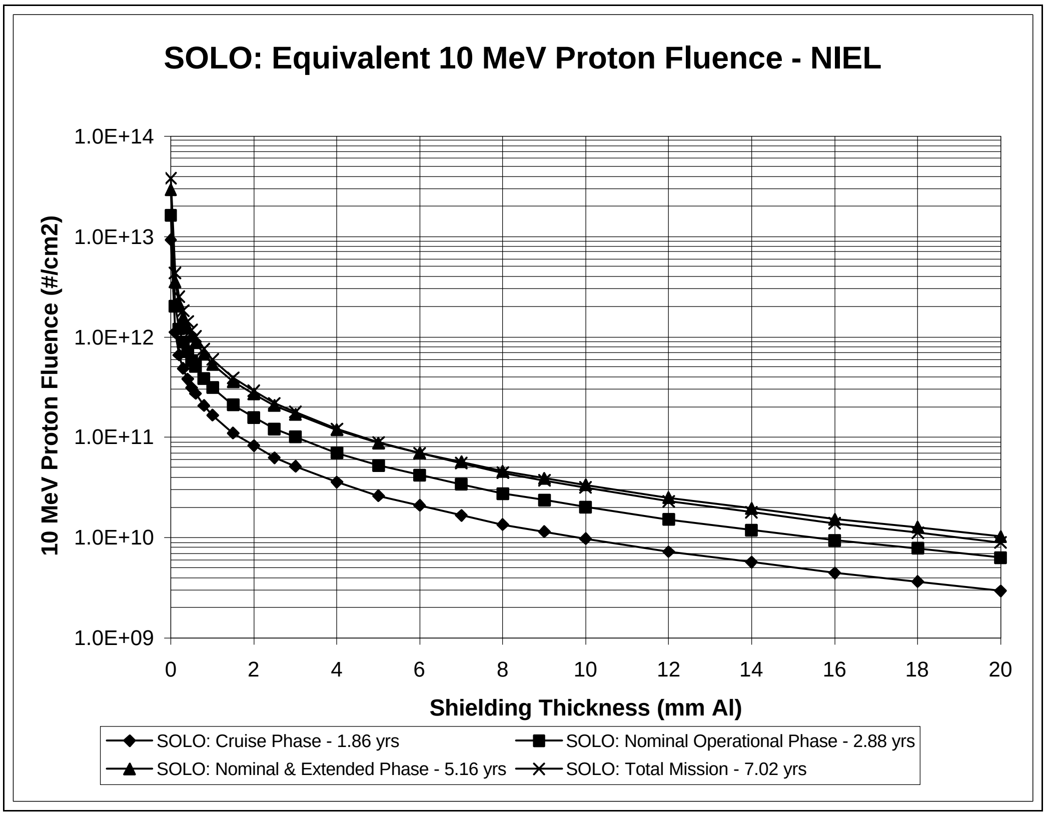

Figure 27-6: Solar Orbiter Equivalent 10 MeV Proton Fluence - NIEL

27.4.6 Interplanetary Cosmic Ray Spectrum

Interplanetary cosmic rays originate outside of the solar system. Fluxes of these particles are low, but because they include heavy energetic ions of elements such as iron, they cause intense ionisation as they pass through matter, and are difficult to shield against. Their primary effect on spacecraft is to induce single event upsets ("bit flips") and latch ups. Two spectra are provided: the 10% worst-case GCR environment with a singly ionised anomalous component (CREME M=3 environment); and the peak flux with worst-case composition from the 10% worst case solar proton event (CREME M=8 environment), scaled by the inverse square of a 0.3 AU heliocentric radius. The "quiet" GCR spectrum is pessimistic, but not greatly so, due to the attenuation of the cosmic rays by interactions with the solar wind.

27.5 Conclusions

The SOLO radiation environment is composed primarily of solar energetic particles. The solar proton environment is considerably harsher than normally expected in Earth orbit. A nominally shielded (4 mm Al.) silicon component is expected to receive a dose of 48kRads over the entire mission. While the total ionising dose is within current engineering standards, c.f. XMM, INTEGRAL, it is expected that CCDs and other opto-electronic components will degrade rapidly in this environment due to bulk /displacement damage effects from energetic protons. Careful consideration of their shielding will be required.

The solar proton fluence spectra have been calculated with a confidence levels that includes a considerable safety margin. No further margin should be required.

A compartive study of the Ulysses data and, e.g. GOES, data should be made to validate the assumption that the solar proton fluence is latitude independent.

27.6 Tabular Data

SOLO: Integral Proton Fluence (#/cm2)				
Energy (MeV)	SOLO: Cruise Phase - 1.86 yrs	SOLO: Nominal Operational Phase - 2.88 yrs	SOLO: Nominal & Extended Phase - 5.16 yrs	SOLO: Total Mission - 7.02 yrs
0.1	1.2E+12	2.1E+12	3.8E+12	4.7E+12
0.5	9.2E+11	1.7E+12	2.9E+12	3.6E+12
1	7.5E+11	1.4E+12	2.4E+12	2.9E+12
2	5.7E+11	1.0E+12	1.8E+12	2.2E+12
3	4.6E+11	8.5E+11	1.5E+12	1.7E+12
4	3.8E+11	7.1E+11	1.2E+12	1.4E+12
5	3.3E+11	6.1E+11	1.1E+12	1.2E+12
6	2.8E+11	5.3E+11	9.2E+11	1.0E+12
8	2.2E+11	4.1E+11	7.1E+11	8.0E+11
10	1.7E+11	3.3E+11	5.7E+11	6.3E+11
12	1.4E+11	2.7E+11	4.7E+11	5.1E+11
15	1.1E+11	2.1E+11	3.6E+11	3.8E+11
17	9.1E+10	1.8E+11	3.0E+11	3.2E+11
20	7.2E+10	1.4E+11	2.4E+11	2.5E+11
25	5.0E+10	9.9E+10	1.7E+11	1.7E+11
30	3.6E+10	7.2E+10	1.2E+11	1.2E+11
35	2.6E+10	5.4E+10	9.0E+10	8.7E+10
40	2.0E+10	4.1E+10	6.8E+10	6.4E+10
45	1.5E+10	3.2E+10	5.2E+10	4.9E+10
50	1.2E+10	2.5E+10	4.1E+10	3.7E+10
60	7.4E+09	1.6E+10	2.6E+10	2.3E+10
70	4.7E+09	1.0E+10	1.7E+10	1.4E+10
80	3.1E+09	6.9E+09	1.1E+10	9.2E+09
90	2.1E+09	4.7E+09	7.6E+09	6.1E+09
100	1.5E+09	3.3E+09	5.3E+09	4.2E+09
120	7.3E+08	1.7E+09	2.6E+09	2.0E+09
140	3.8E+08	8.9E+08	1.4E+09	1.0E+09
160	2.1E+08	4.9E+08	7.6E+08	5.3E+08
180	1.1E+08	2.8E+08	4.3E+08	2.9E+08
200	6.5E+07	1.6E+08	2.5E+08	1.6E+08

Table 27-2: Solar Proton Integral Fluence spectra for the phases of the SOLO mission; units are in particles/cm^2.

SOLO: Dose in Si (Rads)				
Aluminium Absorber Thickness (mm)	SOLO: Cruise Phase - 1.86 yrs	SOLO: Nominal Operational Phase - 2.88 yrs	SOLO: Nominal & Extended Phase - 5.16 yrs	SOLO: Total Mission - 7.02 yrs
0.05	699700.0	1257200.0	2213800.0	2743100.0
0.10	447020.0	810400.0	1421400.0	1729800.0
0.20	269490.0	493810.0	862050.0	1026700.0
0.30	199090.0	367290.0	639280.0	751110.0
0.40	159610.0	295970.0	513990.0	597650.0
0.50	132810.0	247340.0	428720.0	494110.0
0.60	113140.0	211520.0	366020.0	418590.0
0.80	86381.0	162540.0	280460.0	316570.0
1.00	69441.0	131370.0	226140.0	252500.0
1.50	45236.0	86504.0	148200.0	161930.0
2.00	33376.0	64330.0	109830.0	118100.0
2.50	25395.0	49295.0	83901.0	88928.0
3.00	20344.0	39721.0	67432.0	70637.0
4.00	14036.0	27682.0	46788.0	48027.0
5.00	10383.0	20645.0	34770.0	35106.0
6.00	8001.5	16025.0	26903.0	26769.0
7.00	6342.1	12783.0	21401.0	21021.0
8.00	5214.5	10569.0	17651.0	17142.0
9.00	4257.8	8678.0	14458.0	13883.0
10.00	3578.4	7328.5	12183.0	11586.0
12.00	2672.5	5520.0	9142.7	8546.3
14.00	2019.7	4204.9	6940.2	6384.1
16.00	1569.3	3290.6	5414.1	4908.9
18.00	1277.4	2694.8	4422.0	3960.3
20.00	1067.6	2264.7	3707.3	3282.8

Table 27-3: Total Solar Proton Dose (Rads) in an infinitesimal Silicon target with a spherical aluminium shield for the SOLO mission phases.

SOLO: (GaAs) -VOC 1 MeV Eq. Fluence (/cm2)				
Depth (microns)	SOLO: Cruise Phase - 1.86 yrs	SOLO: Nominal Operational Phase - 2.88 yrs	SOLO: Nominal & Extended Phase - 5.16 yrs	SOLO: Total Mission - 7.02 yrs
0	7.0E+15	1.2E+16	2.2E+16	2.9E+16
25	1.2E+15	2.1E+15	3.7E+15	4.7E+15
76	5.8E+14	1.1E+15	1.9E+15	2.3E+15
152	3.5E+14	6.4E+14	1.1E+15	1.3E+15
305	2.0E+14	3.7E+14	6.4E+14	7.4E+14
509	1.3E+14	2.4E+14	4.2E+14	4.7E+14

SOLO: (GaAs) -PMAX 1 MeV Eq. Fluence (/cm2)				
Depth (microns)	SOLO: Cruise Phase - 1.86 yrs	SOLO: Nominal Operational Phase - 2.88 yrs	SOLO: Nominal & Extended Phase - 5.16 yrs	SOLO: Total Mission - 7.02 yrs
0	5.0E+15	8.7E+15	1.6E+16	2.1E+16
25	8.4E+14	1.5E+15	2.7E+15	3.3E+15
76	4.2E+14	7.6E+14	1.3E+15	1.6E+15
152	2.5E+14	4.6E+14	8.0E+14	9.5E+14
305	1.4E+14	2.6E+14	4.6E+14	5.3E+14
509	9.2E+13	1.7E+14	3.0E+14	3.4E+14

SOLO: (GaAs) -ISC 1 MeV Eq. Fluence (/cm2)				
Depth (microns)	SOLO: Cruise Phase - 1.86 yrs	SOLO: Nominal Operational Phase - 2.88 yrs	SOLO: Nominal & Extended Phase - 5.16 yrs	SOLO: Total Mission - 7.02 yrs
0	3.0E+15	5.2E+15	9.4E+15	1.2E+16
25	4.2E+14	7.6E+14	1.3E+15	1.7E+15
76	2.0E+14	3.5E+14	6.2E+14	7.6E+14
152	1.1E+14	2.0E+14	3.6E+14	4.3E+14
305	6.0E+13	1.1E+14	1.9E+14	2.2E+14
509	3.7E+13	7.0E+13	1.2E+14	1.4E+14

Table 27-4: Silicon Solar Cell fluence of equivalent 1 MeV electrons for the SOLO mission phases. Top: P_{MAX}-V_{OC}, Bottom: I_{SC}.

SOLO: 10 MeV Equiv. Proton Fluence				
Aluminium Absorber Thickness (mm)	SOLO: Cruise Phase - 1.86 yrs	SOLO: Nominal Operational Phase - 2.88 yrs	SOLO: Nominal & Extended Phase - 5.16 yrs	SOLO: Total Mission - 7.02 yrs
0.001	9.3E+12	1.6E+13	2.9E+13	3.8E+13
0.1	1.1E+12	2.0E+12	3.6E+12	4.3E+12
0.2	6.6E+11	1.2E+12	2.1E+12	2.5E+12
0.3	4.8E+11	8.9E+11	1.6E+12	1.8E+12
0.4	3.8E+11	7.1E+11	1.2E+12	1.4E+12
0.5	3.1E+11	5.8E+11	1.0E+12	1.2E+12
0.6	2.7E+11	5.1E+11	8.8E+11	1.0E+12
0.8	2.1E+11	3.9E+11	6.7E+11	7.5E+11
1	1.6E+11	3.1E+11	5.4E+11	6.0E+11
1.5	1.1E+11	2.1E+11	3.6E+11	3.9E+11
2	8.2E+10	1.6E+11	2.7E+11	2.9E+11
2.5	6.3E+10	1.2E+11	2.1E+11	2.2E+11
3	5.1E+10	1.0E+11	1.7E+11	1.8E+11
4	3.6E+10	7.0E+10	1.2E+11	1.2E+11
5	2.6E+10	5.2E+10	8.8E+10	8.8E+10
6	2.1E+10	4.2E+10	7.0E+10	6.9E+10
7	1.7E+10	3.4E+10	5.7E+10	5.5E+10
8	1.4E+10	2.8E+10	4.6E+10	4.4E+10
9	1.2E+10	2.4E+10	3.9E+10	3.7E+10
10	9.8E+09	2.0E+10	3.4E+10	3.2E+10
12	7.3E+09	1.5E+10	2.5E+10	2.3E+10
14	5.7E+09	1.2E+10	2.0E+10	1.8E+10
16	4.4E+09	9.4E+09	1.5E+10	1.4E+10
18	3.7E+09	7.8E+09	1.3E+10	1.1E+10
20	2.9E+09	6.3E+09	1.0E+10	9.0E+09

Table 27-5: Non-Ionising dose: equivalent 10 MeV proton fluence (#/cm^2) as a function of spherical shell shielding thickness.